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GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

STUDY OF PRESSURIZATION SYSTEMS FOR LIQUID PROPULSION ROCKET ENGINES (U)

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(Final)

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The Space Propulsion Division
of the Liquid Rocket Plant
Azusa, California

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AEROJET-GENERAL CORPORATION

Azusa, California

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FOREWORD

This report is submitted in fulfillment of NASA Goddard Space Flight Center Contract No. NAS 5-1108. It includes, together with the material which will be published in Report No. 2334, the work previously published in quarterly reports on the study of pressurization systems for liquid-propellant rocket engines. The following tasks are covered:

Task I	Data compilation and component operating characteristics; assembly of data generated for Report No. 2334 in three volumes
Task II	System selection and design
Task III	System fabrication and testing

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I. PROGRAM SUMMARY

This final report presents the results of a comprehensive study performed under NASA Goddard Space Flight Center Contract No. NAS 5-1108 to devise a selection technique for propellant pressurization systems. As a result of the study, the likely propellant pressurization systems for advanced space vehicles were determined, and a method of preliminary design and selection of the most suitable of these systems for any specific space mission was provided.

A. PROGRAM OBJECTIVES

The objectives of this study were as follows: (1) to generate design data on propellant pressurization system components, (2) to present a method of combining these components into likely systems, (3) to provide a technique for comparing and selecting the most suitable pressurization system for use on specific space missions.

This report summarizes the results of the entire program. System analysis and evaluation technique is presented along with the method of combining components into a working propellant pressurization. Design data were prepared for comparing and selecting the most suitable pressurization system for use on space missions. Specific component design data were summarized for this use. A self-contained design guide will be written, and may be expanded while this final report will not. It will therefore be issued as a separate report, Aerojet Report No. 2334 in three volumes. Some of the material contained in this final report, No. 2335, therefore has also been placed in Report No. 2334. Sections of this report also appearing in Report No. 2334 have been so designated.

B. SCOPE

All of the commonly used propellant pressurization systems available for use on current space vehicles, excepting mechanical pumping systems, have been

considered. Some novel pressurization concepts, e.g., a regenerative jet pump, also were investigated.

The space vehicles on which these systems will be used may be manned or unmanned. Another study program "Research Study to Determine Propulsion Requirements and Systems for Space Missions," Contract NAS 5-915, is also being conducted for NASA by Aerojet-General Corporation. The missions, defined by this study, include lunar and planetary orbits and lunar landings. These missions were used in part in defining the propellant pressurization requirements for this program.

Pressurization system components were investigated on the basis of performance, reliability, weight, size, space environmental effects, cost, and material compatibility. Design and evaluation data were compiled on these bases. Further, the pressurization systems studied were based upon, but not limited to, the following propellant combinations.

Cryogenic	Storable
LO_2/LH_2	ClF_3/Hydrazoid
LF_2/LH_2	N_2O_4/N_2H_4
OF_2/LH_2	N_2O_4/Aerozine-50
	N_2H_4/Monopropellant

C. DISCUSSION

To accomplish the program objectives, without duplication of work already accomplished in the propellant pressurization system field, the study was divided into the following four tasks:

Task I	Data compilation and component operational characteristics
Task II	System selection and design
Task III	System fabrication and testing

The results of Task I meet objective (1), i.e., compiling design data on propellant pressurization system components. The results will be presented in Report No. 2334.

Task II fulfills objectives (2) and (3). A method for generating new pressurization systems and a method for selecting the most promising system are presented in this report. These sections will be repeated with the publication of the design data compiled under Task I.

Task III consists of the fabrication and testing of components for one pressurization system to provide design data for objective (1).

The study began with the collection of data from Government and private industry engaged in the field of liquid rocket propulsion and pressurization systems. These data were analyzed, and parametric performance and evaluation equations were derived. These equations will appear in Report No. 2334. With a complete set of data on the system components available, a method of combining these components into different system combinations was prepared. A technique for evaluating and comparing these systems was presented and trial missions were chosen to demonstrate the use of the method. One of the systems selected during the comparison was fabricated and tested, and the test results were used to verify the theoretical predictions. The test results appear in this report. Finally, the design data were prepared in a form for publication as a handbook.

The literature searches, vendor contact, and discussions with the NASA program technical director were used to compile information on propellant pressurization systems and components presently in use, or proposed for use in the future. The technical information centers used in the literature search include Aerojet-General Libraries, Interlibrary loan agencies, ASTIA and LPIA.

A method of combining the components described into propellant pressurization systems was then devised. A modified morphological approach was employed after the components were grouped by function. Selecting a component from each function rather than at random greatly reduces the number of unlikely combinations required to be evaluated.

Component performance and evaluation, which comprised the major effort in the study, is described in detail and the characteristics are expressed in parametric equations. Design data include, e.g., sizing of flow passages, determination of material selection and dimensions, calculations of spring rates, and operating temperature and pressure limits.

The evaluation was used to determine the overall rating of each component combination. The method of rating makes use of a series of rating curves to convert the evaluation factors of each system to a point rating. The higher the point rating, the more suitable the system for the intended mission. To demonstrate the use of the selection method, ratings were performed on example missions. The missions chosen by NASA for demonstration were the lunar landing, lift-off and return, and the Mars orbit. The selection method shows main tank injections (MTI) to be most suitable for one of these missions. The MTI system was designed utilizing the parametric equations developed previously. A model of the system was fabricated and tested. Where possible, existing components and test equipment were utilized. Instrumentation was used to record component performance and system process.

This final report, Aerojet Report No. 2335, summarizes the results of the entire program including recommendations and conclusions.

D. DESIGN HANDBOOK

A design handbook, Aerojet Report No. 2334, will present component performance and evaluation. Data, equations, and graphs will be presented which shall enable the user to design and evaluate many types of propellant pressurization systems including hybrid and redundant systems. Evaluation data, for use in the comparative selection technique, will include weight, volume, reliability, and cost.

The handbook also will include information on the selection and compatibility of component materials and the effects of space environment on them. A brief résumé of the physical and thermodynamic properties of pressurants and propellants will be presented. Many component combinations, both used and proposed, are reviewed. A section of the handbook applies to the procedure utilized in designing a propellant pressurization system to meet particular specifications.

II. MORPHOLOGICAL APPROACH*

A. CONCEPT OF THE MORPHOLOGICAL APPROACH

The selection of an optimum pressurization system is dependent both on the ability to select systems to evaluate and on the method used to establish the relative merit of the systems selected. In this section the first aspect, that of determining possible systems to evaluate, will be discussed.

The ultimate in widening the scope of the system considered for any mission would be the morphological approach. The concepts underlying this approach are as follows:

1. Establish the list of components of which any pressurization system may be composed.

2. Generate all combinations which can be formed by the component array.

3. Generate all permutations which can be formed by the component combinations.

It is, therefore, a systematic procedure which will generate a vast number of candidate pressurization systems. For the components considered, the system has the potential of generating all possible pressurization systems for any mission. The difficulty encountered by this approach is that if enough components are included to make the method useful, more candidate systems are generated than can possibly be evaluated. This causes a great deal of time to be spent culling out obviously inoperative "systems." Some method of avoiding the vast number of inoperative systems must be implemented before this approach is practical.

B. MODIFIED MORPHOLOGICAL APPROACH

Intuitively, one can recognize that all component groupings which could truly qualify as pressurization systems are subject to further limitations. These are functional operations which must be performed by the components in order for the system to "pressurize" at all. Thus we wish to restrict ourselves to the component associations which are capable of delivering pressurization media.

*This section will also appear in Aerojet Report No. 2334.

In the process of developing this approach, grouping of the components that perform similar functions was found to be highly advantageous. This modified morphological approach reduces the number of possible combinations and leads to a selection technique that is more easily handled.

1. Modified Morphological Approach

The application of this modification is accomplished as follows:

a. Establish the ordered set of performance functions of which a generalized pressurization system is composed.

b. Establish the list of components which are to be considered in each functional set.

c. Generate all possible component combinations which can be formed by placing components only in the positions reserved for the functional sets to which they belong.

d. Examine the resulting systems for practicality and component compatibility.

2. Component Categories

It was found that all pressurization system components could be grouped into six ordered-function categories, Figure II-1. Any number of components from none to several, may be selected from each category. The six categories are as follows:

a. Energy Supplies

This category includes all primary energy sources and their containers. High-pressure stored gases, liquid-propellant gas generators, solid-propellant gas generators, thrust-chamber heat exchangers, and batteries are covered in this section. The properties of gases and products of combustion will be included together with the analysis of associated flow processes such as the use of gas from a high-pressure storage container.

b. Initiators/Terminators

This section will cover the design of devices for commencing or terminating system operations such as igniters, electrical switches, solenoid-operated valves, explosive valves, and burst diaphragms. The size and weight of these items is dependent upon the energy demand and the operating conditions.

c. Charge and Recharge Connectors

Electrical connectors and fluid-line disconnects will be discussed under this heading. Design and evaluation data will be given in terms of the desired charging rate.

d. System Controls

In most propellant pressurization systems the energy-converting component is the "heart" of the system.

The task of maintaining a constant energy supply under varying load conditions often requires a complex component. This section will cover the design of pressure regulators and orifices.

e. Transmission Systems

The energy required to feed propellants to the engine must be transmitted from the supply to the propellant by one or more "conductors." Electric sources require wiring, mechanical sources require gears, and pressure sources require tubing to transmit energy. The transmission components will be described as a function of the energy supply rate.

f. Safety Devices

Most propellant feed systems employ safety devices to increase reliability and reduce operating hazards. Check valves prevent interflow betweeen propellant tanks, electrical relays and relief valves prevent overload, and bladders prevent hot gases from coming in direct contact with the propellant. The design of safety devices and reasons for their use will be presented under this category.

C. EXAMPLE OF COMPONENT COMBINATIONS

The modified morphological approach, described above, was employed to select 16 workable component combinations. One or two components were selected from each of the performance categories described above, and the tabulation is shown in Table II-I.* These 16 systems, which are used as the examples of the evaluation technique, represent but a small percentage of the workable systems which could be formed using this approach.

In an attempt to maintain the objective of the program for unbiased system evaluation, the systems were formed without consideration of a particular mission. Every component being evaluated in this study is included in one or more of the systems. Schematic diagrams of the 16 systems have been prepared and are shown in Figures II-2 through II-17.

Component Combinations 8, 15, and 16 show that at least three basic types of hybrid propellant pressurization systems can be created. Component Combination 8 employs two energy supplies (high-pressure gas and a heat exchanger) functioning simultaneously to expel the propellant. Combination 15 employs two energy supplies, one for expelling each propellant. In Component Combination 16 two energy supplies are used consecutively, one being employed after depletion of the other. Component Combination 14 is an even more complex hybrid incorporating the features of both Combinations 8 and 10.

The formation of novel hybrid propellant pressurization systems appears to be a very promising area for the application of the modified morphological approach. With anticipated space missions being composed of several maneuvers, it is possible that propellant pressurization systems powered by two or more energy supplies, each functioning when it best suits the maneuver, could prove to be the lightest in weight or the most reliable.

D. FUTURE POTENTIAL OF THE MORPHOLOGICAL APPROACH

Due to the large number of possible component combinations, an approach such as a modified morphological development appears to be the only

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*Tables and figures pertaining to a particular section may be found at the end of that section.

practical technique that will permit all the alternative systems to be appraised. Presently, the designer tends to limit himself to variations of the relatively few systems with which he is familiar, and does not make full utilization of the components available to him. This is a result of the designer being discouraged by the number of components available and the geometric nature of the combining process.

This situation leads to the possible introduction of computer usage as a practical and expedient method of both generating and appraising a large number of possible systems. This would basically increase the breadth of the designers investigation and consequently permit a more thorough analysis of the possible systems.

The usefulness of computers in this particular application would depend primarily on the extent that factors, such as component compatibility, mission limits, and a weighting for the preference of proven systems over new untested systems could be incorporated into the program.

Moreover, once such a program is developed, it would have the potential of being used by designers regardless of their project or company affiliation and/or by a project administrator for the appraisal of pressurization systems that must meet certain design features. The future of such an undertaking is unlimited. Over a period of time refinements would evolve that would continually increase the sophistication of the methodology and, therefore, the program's overall usefulness.

TABLE II-1

COMPONENT COMBINATIONS

	1	2	3	4	5	6	7	8
ENERGY SUPPLY	High Press. Gas	High Press. Gas	High Press. Gas	Preconditioned Propellant	Battery Vaporized Propellant	Propellant High Press.Gas	Propellant High Press.Gas	High Press. Gas TCA Heat Xchgr.
INITIATORS/TERMINATORS	Solenoid Valve	Squib Valve	Solenoid Valve	None	Electrical Switch	Solenoid Valve	Solenoid Valve	Solenoid Valve
CHARGE & RECHARGE CONNECTORS	Pressure Line Disconnect	Pressure Line Disconnect	Pressure Line Disconnect	Propellant Line Disconnect	Propellant Line Disconnect	Pressure Line Disconnect Propellant Line Disconnect	Pressure Line Disconnect Propellant Line Disconnect	Pressure Line Disconnect
SYSTEM CONTROL	Press Regulator	Orifice	Orifice	None	Pressure Regulator Pressure Switch	Press. Regulator	Press. Regulator	Press. Regulator
TRANSMISSION SYSTEM	Tubing	Tubing	Tubing	Tubing	Tubing/Wires	Tubing	Tubing	Tubing
SAFETY DEVICES	Check Valves Relief Valves	Check Valves Relief Valves	Check Valves Relief Valves	Relief Valve	Relief Valve	Check Valve Relief Valve	Relief Valve Filter	Check Valve Relief Valve
TYPE OF SYSTEM	Stored Gas	Stored Gas	Stored Gas	Vāpak	Secondary Vāpak	MTI	MTI	Heated Stored Gas

TABLE II-1 (cont.)

COMPONENT COMBINATIONS (CONT.)

	9	10	11	12	13	14	15	16
ENERGY SUPPLY	Battery	LGG High Pressure Gas	SGG	SGG	High Pressure Gas TCA Heat Exchgr.	High Pressure LGG Heat Exchanger	High Press. Gas Evaporated Fuel Heat Exchanger	High Pressure Gas SGG
INITIATOR/TERMINATOR	Electrical	Solenoid Valve	Ignitor Burst Diaphragm	Ignitor Burst Diaphragm	Solenoid Valve	Solenoid Valve	Solenoid Valve 3-Way Solenoid Valve	Solenoid Valve Igniter
CHARGE & RECHARGE CONNECTORS	None	Press. Line Disconnect Propellant Line Disconnect	None	None	Propellant Line Disconnects Pressure Line Disconnect	Propellant Line Disconnects	Pressure Line Disconnect Propellant Line Disconnect	Pressure Line Disconnect
SYSTEM CONTROL	Electric Motor	Pressure Regulator	Orifice	Orifice	Jet Pump	Variable Orifice Pressure Regulator	Pressure Regulator	Pressure Regulator Pressure Switch
TRANSMISSION	Wires, Gears Bellows	Tubing	Tubing	Tubing	Tubing	Tubing	Tubing	Tubing, wires
SAFETY DEVICES	Check Valve	Check Valves Relief Valves	Filter Check Valves Relief Valve	Filter Check Valve Relief Valve Bladder	Relief Valve	Relief Valve	Check Valve Relief Valve	Check Valves Relief Valves
TYPE OF SYSTEM	Mechanical Bellows	Liquid Propellant Gas Generator	Solid Propellant Gas Generator	Solid Propellant Gas Generator	Jet Pump	Hybrid LGG/Line Heated Stored Gas	Hybrid Line Heated Stored Gas/EP	Hybrid Stored Gas/SGG

MORPHOLOGICAL OUTLINE

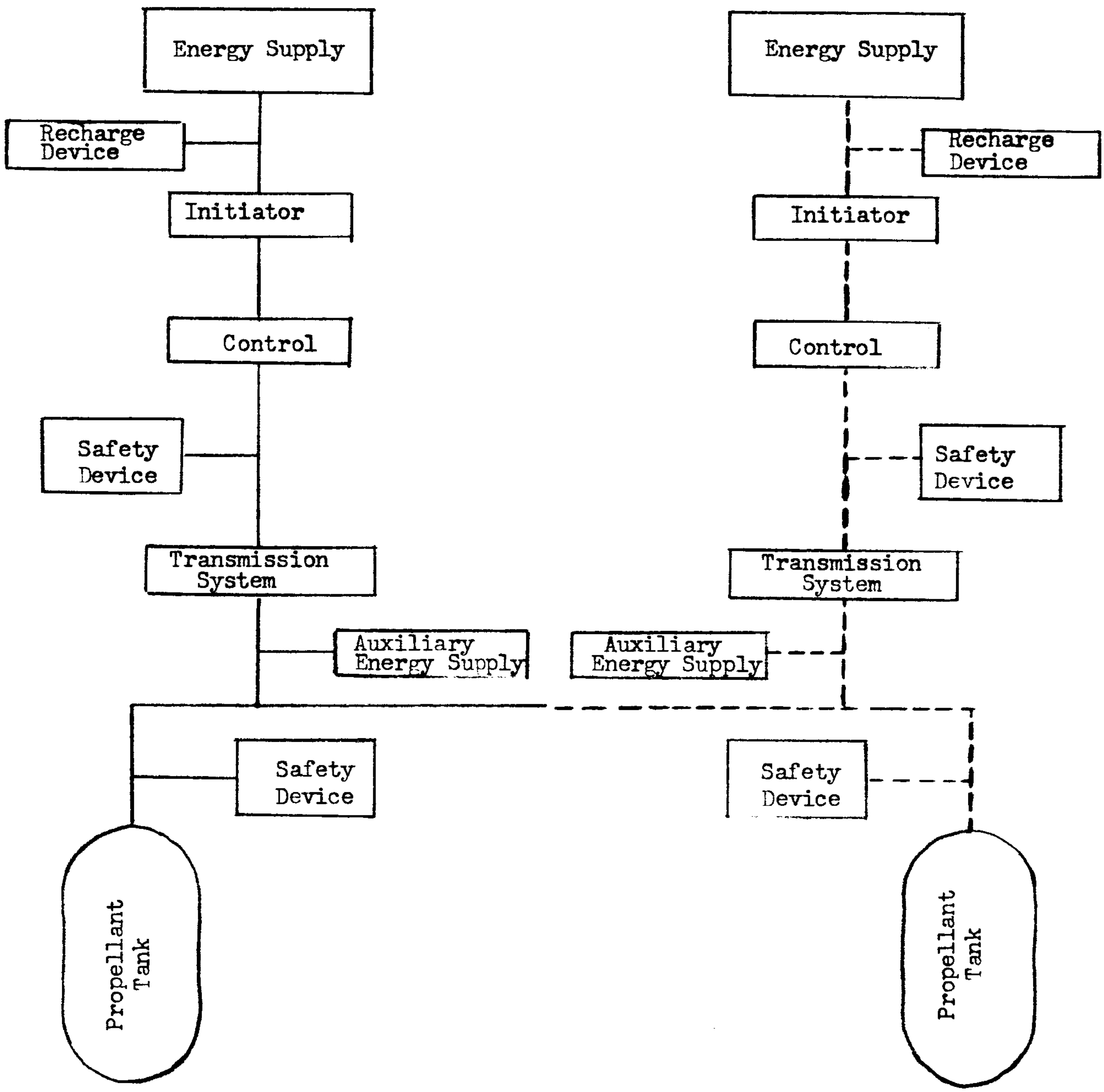
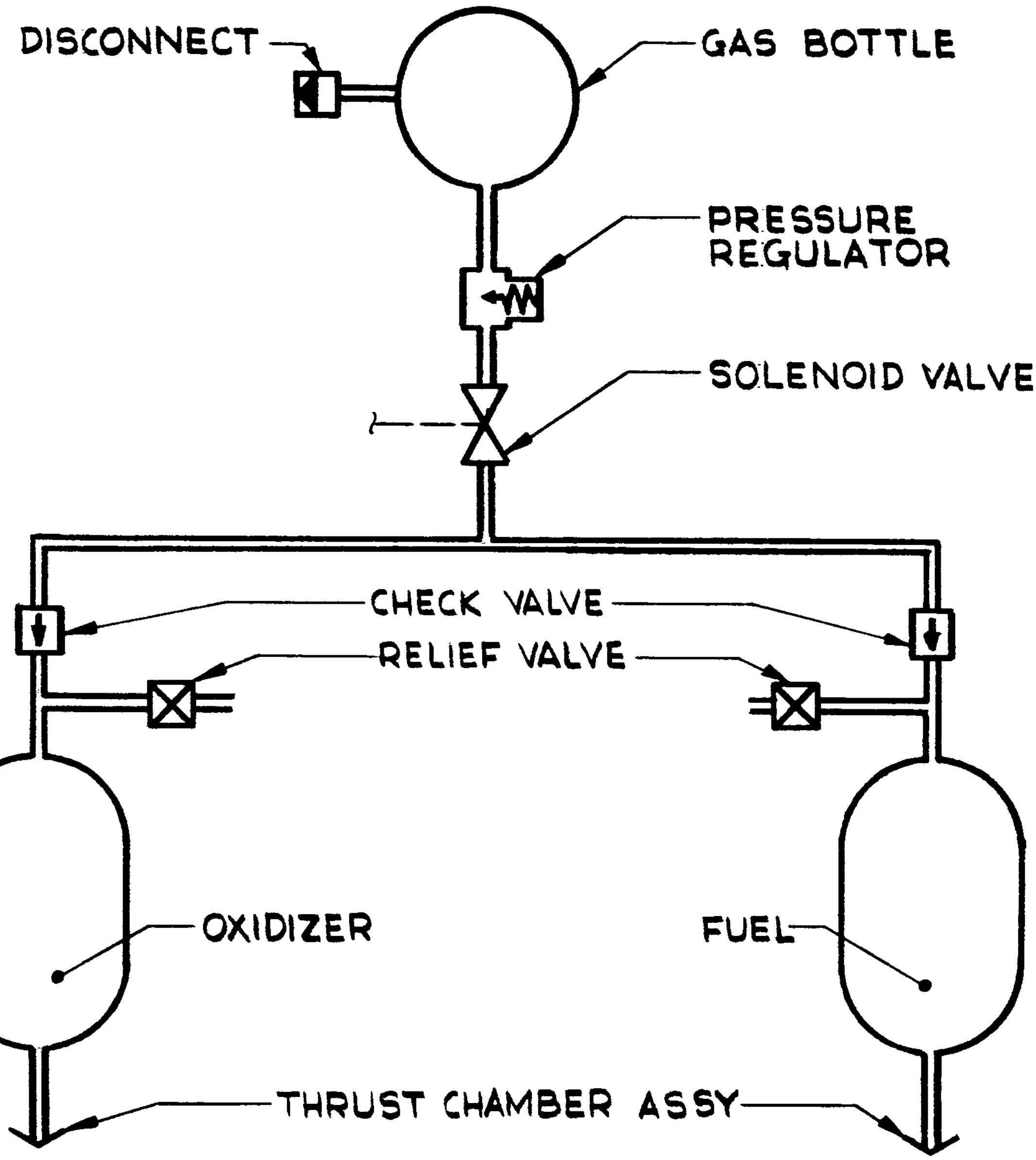


Figure II-1



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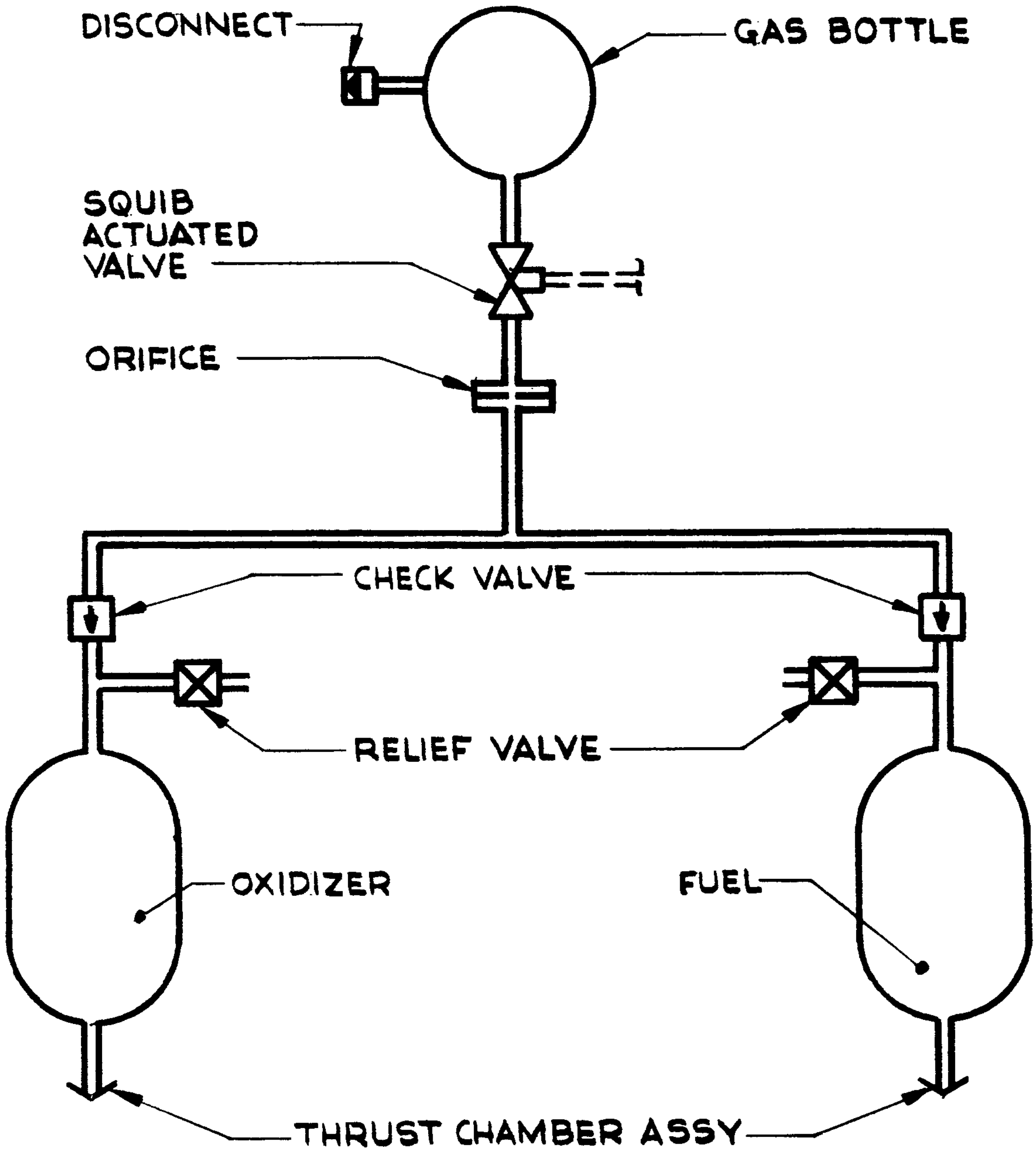

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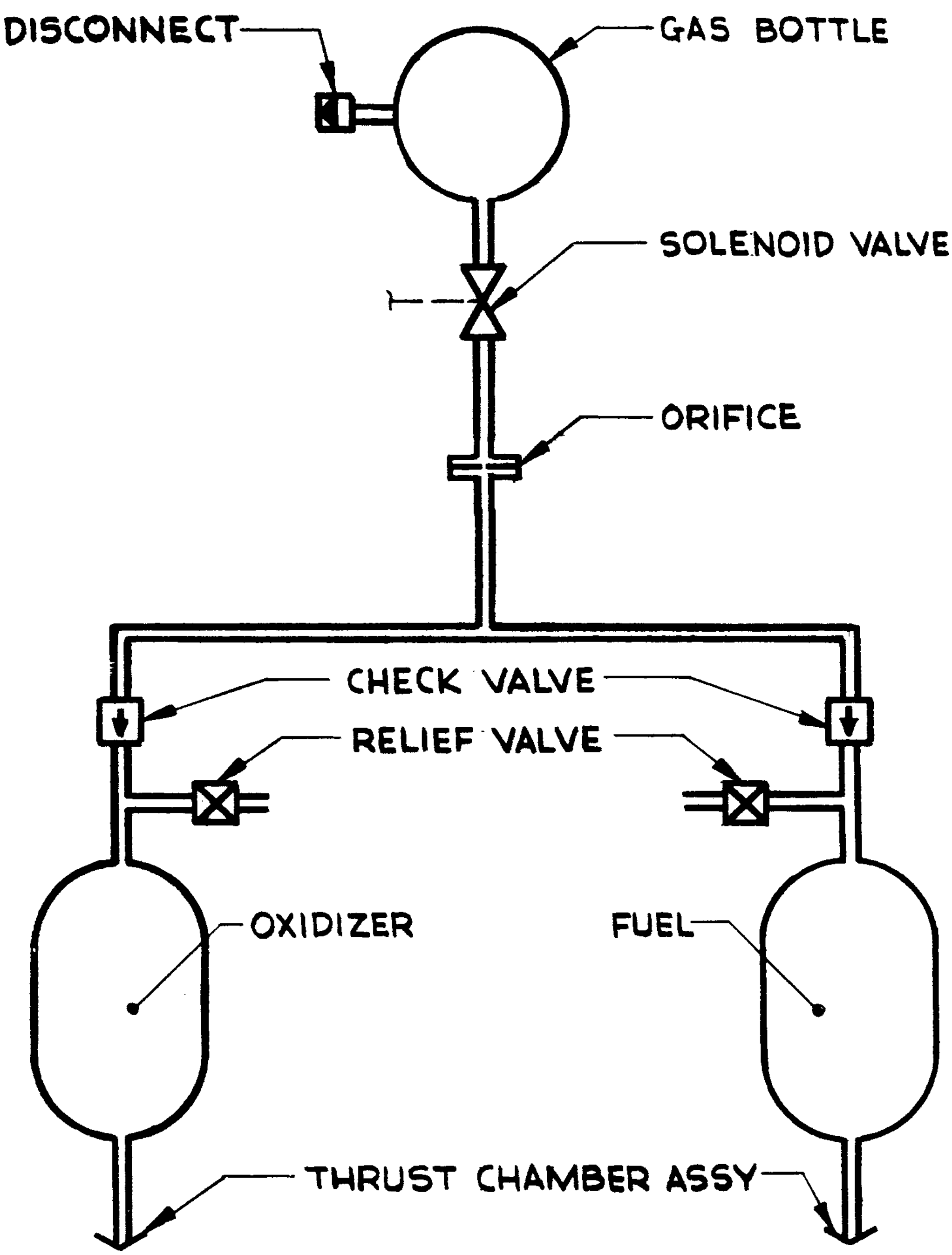

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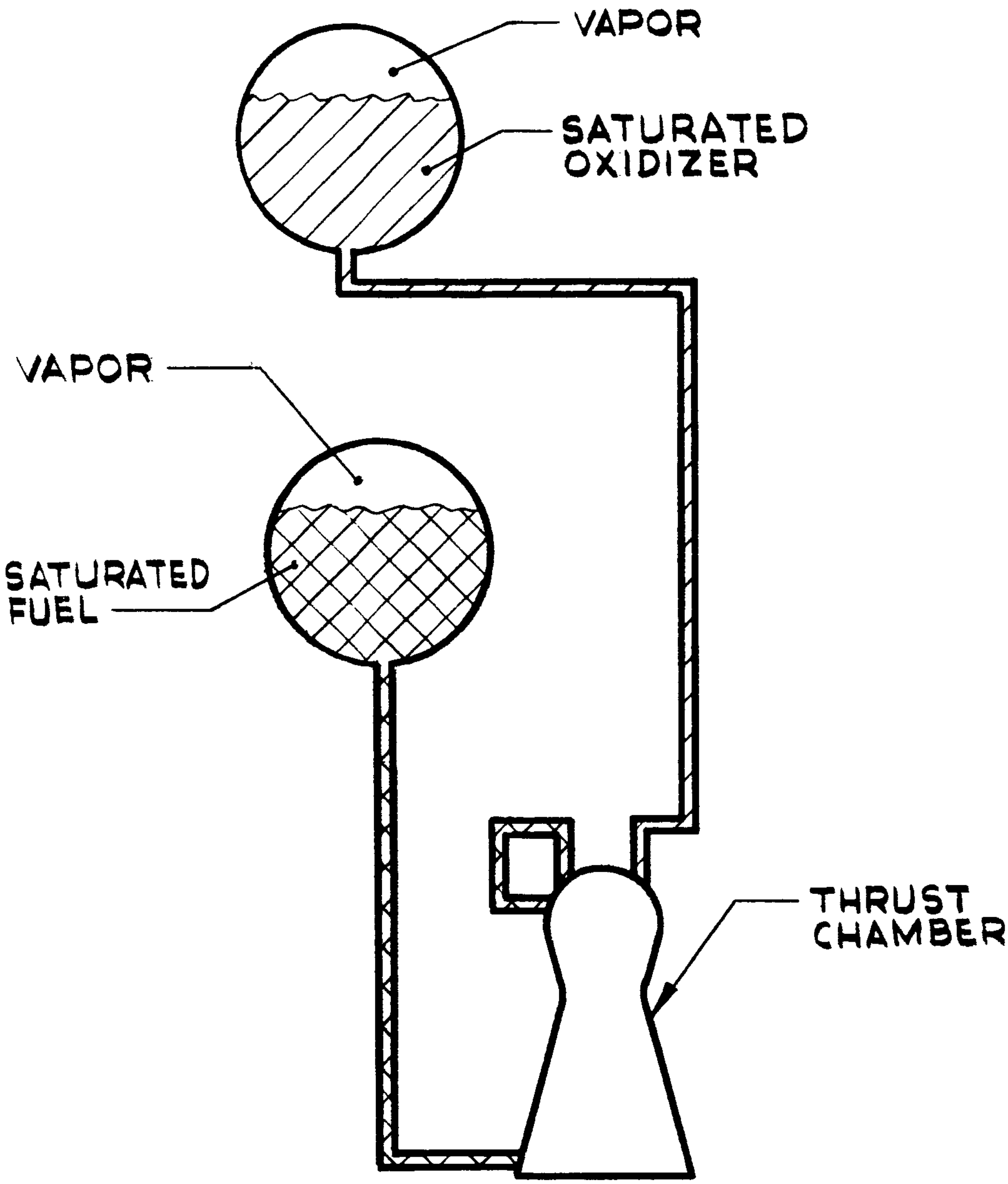

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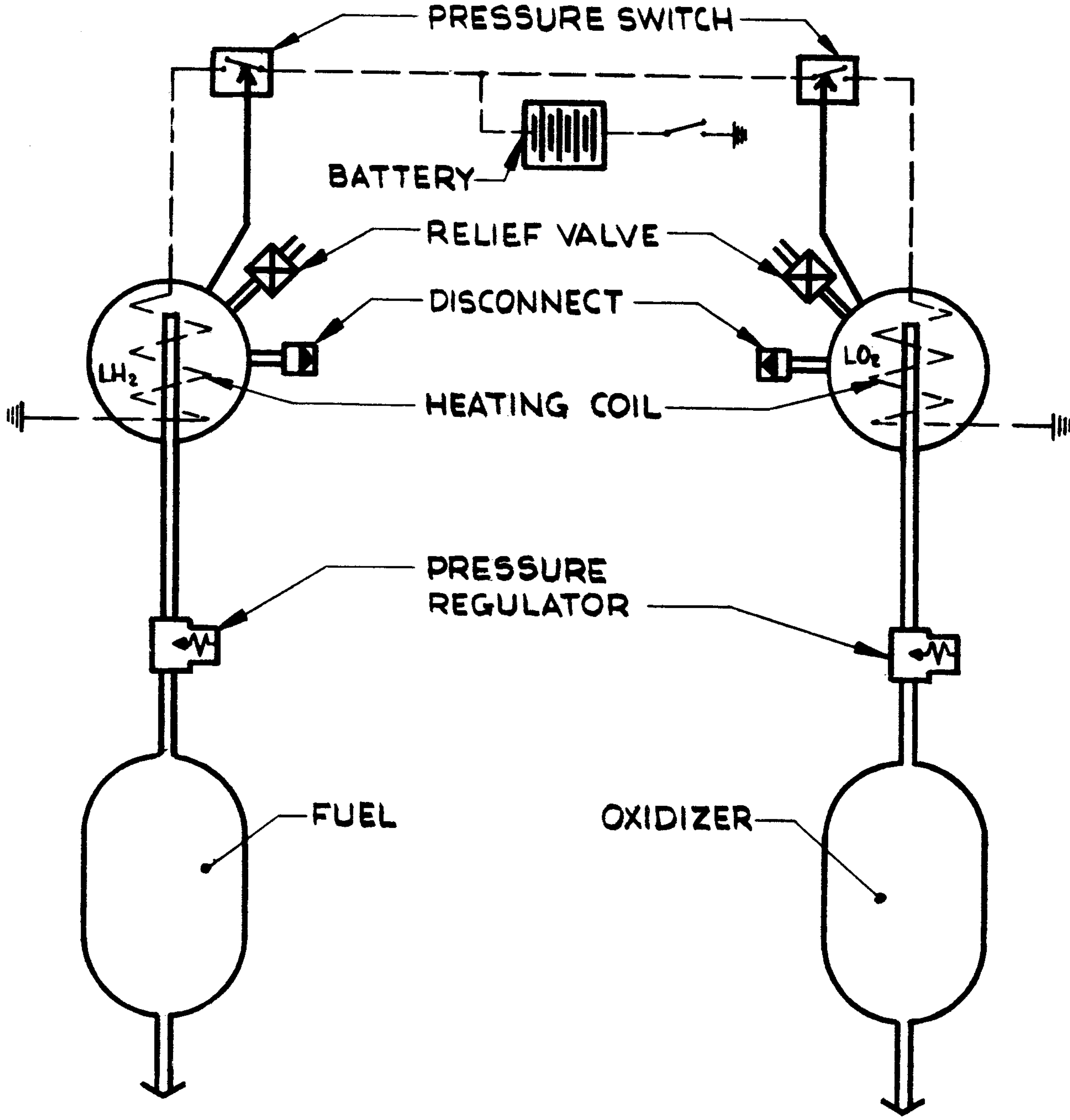

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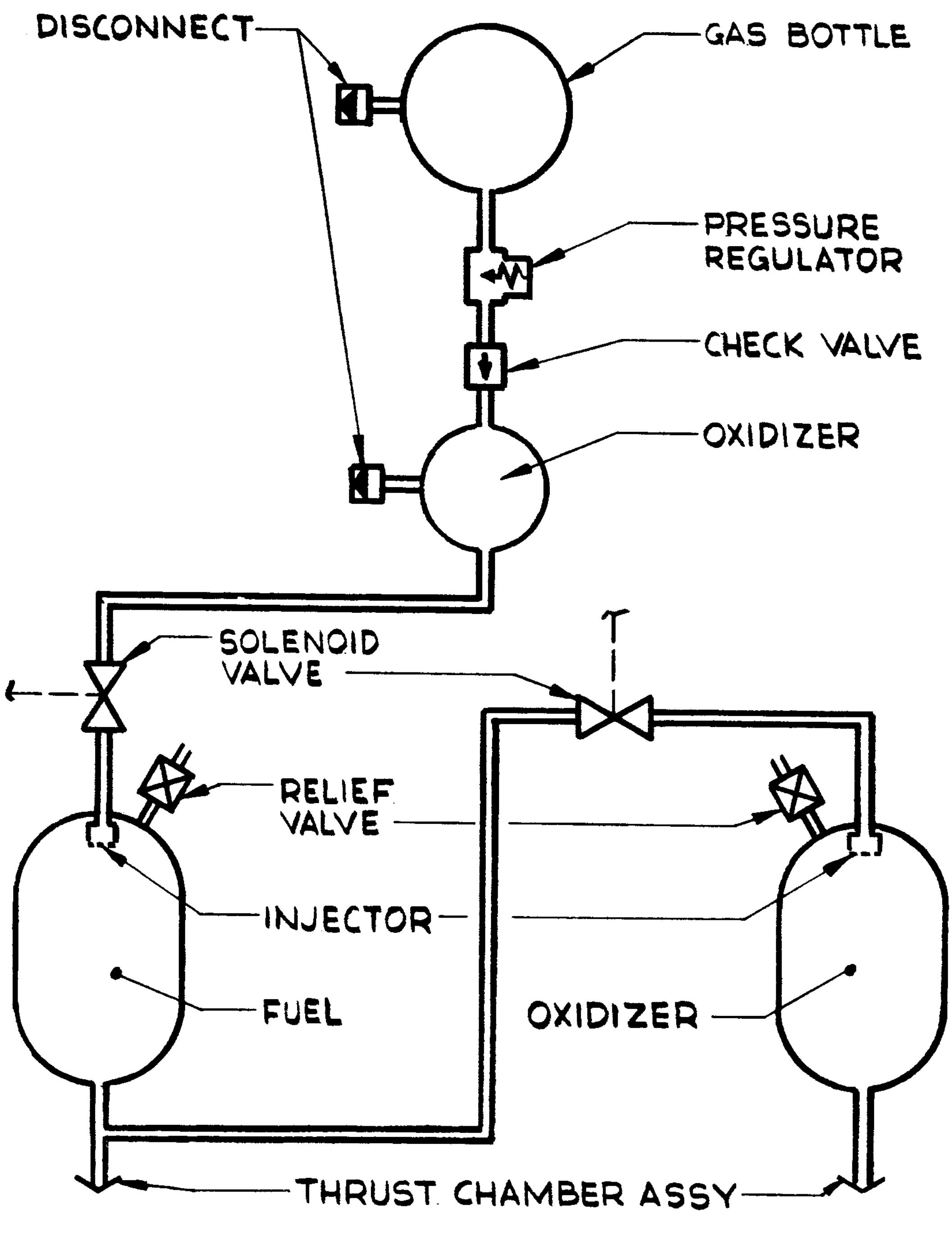

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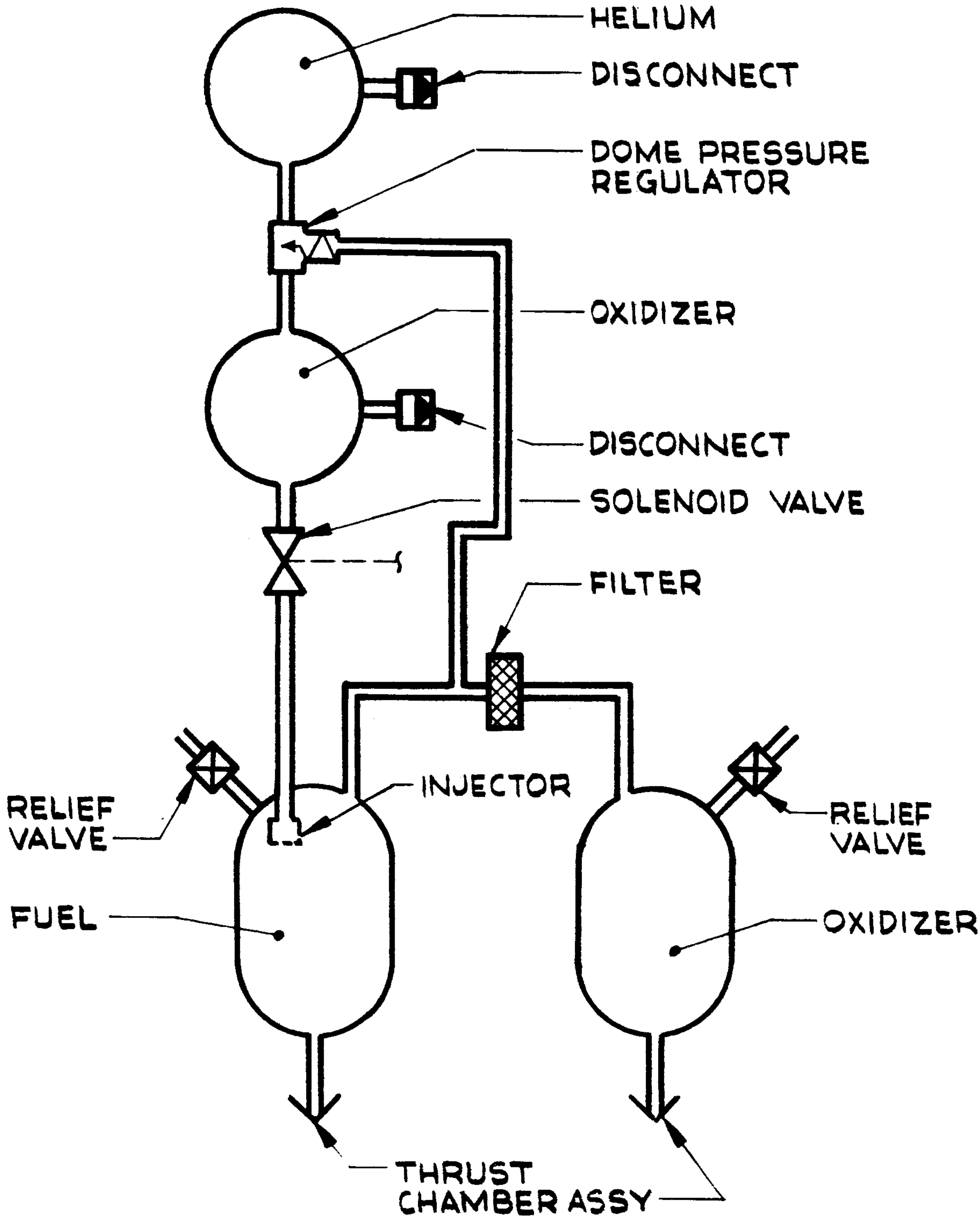

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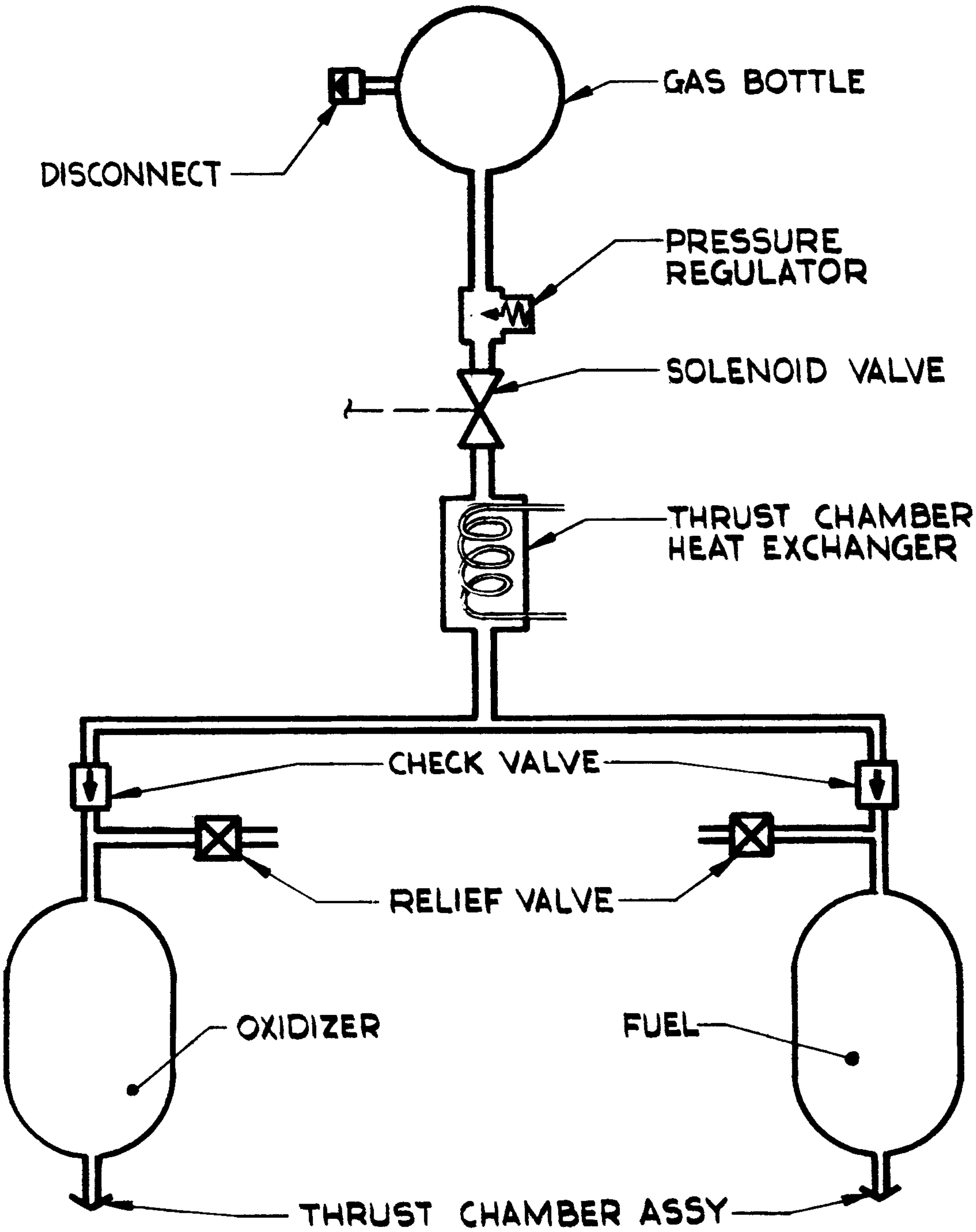

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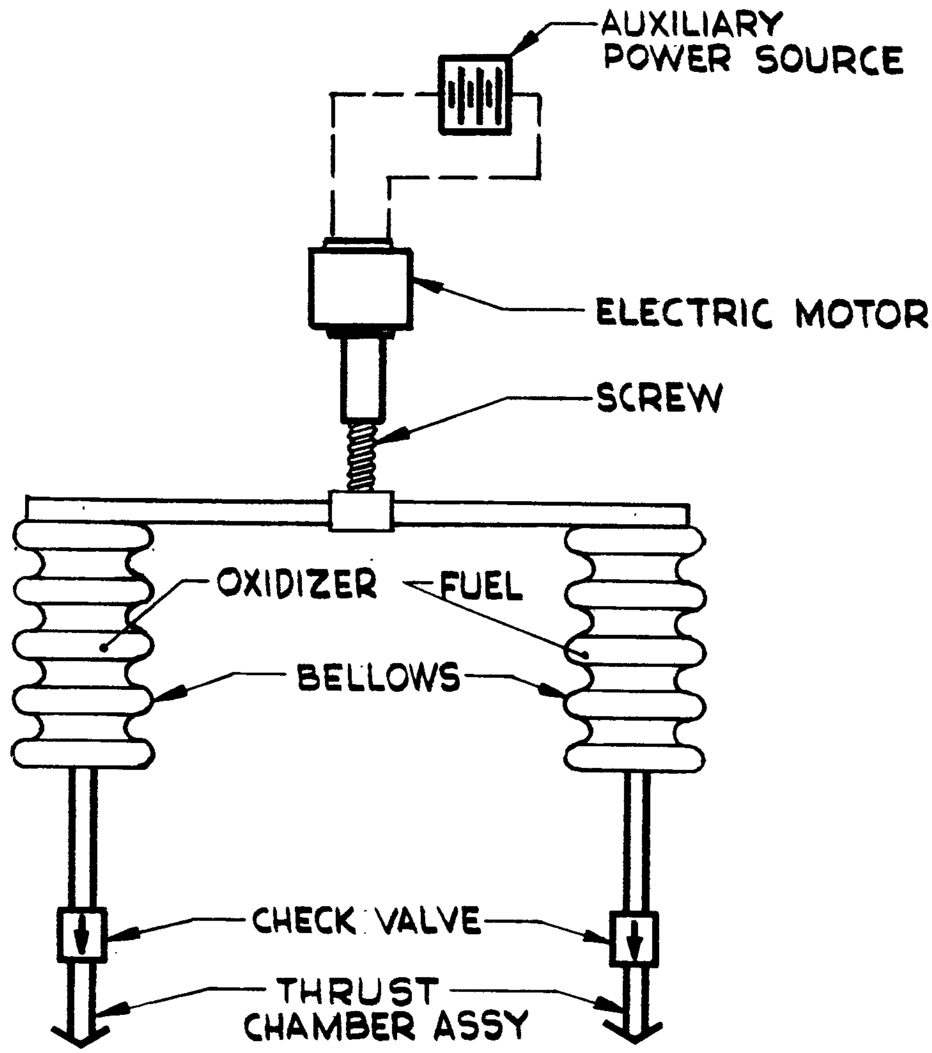

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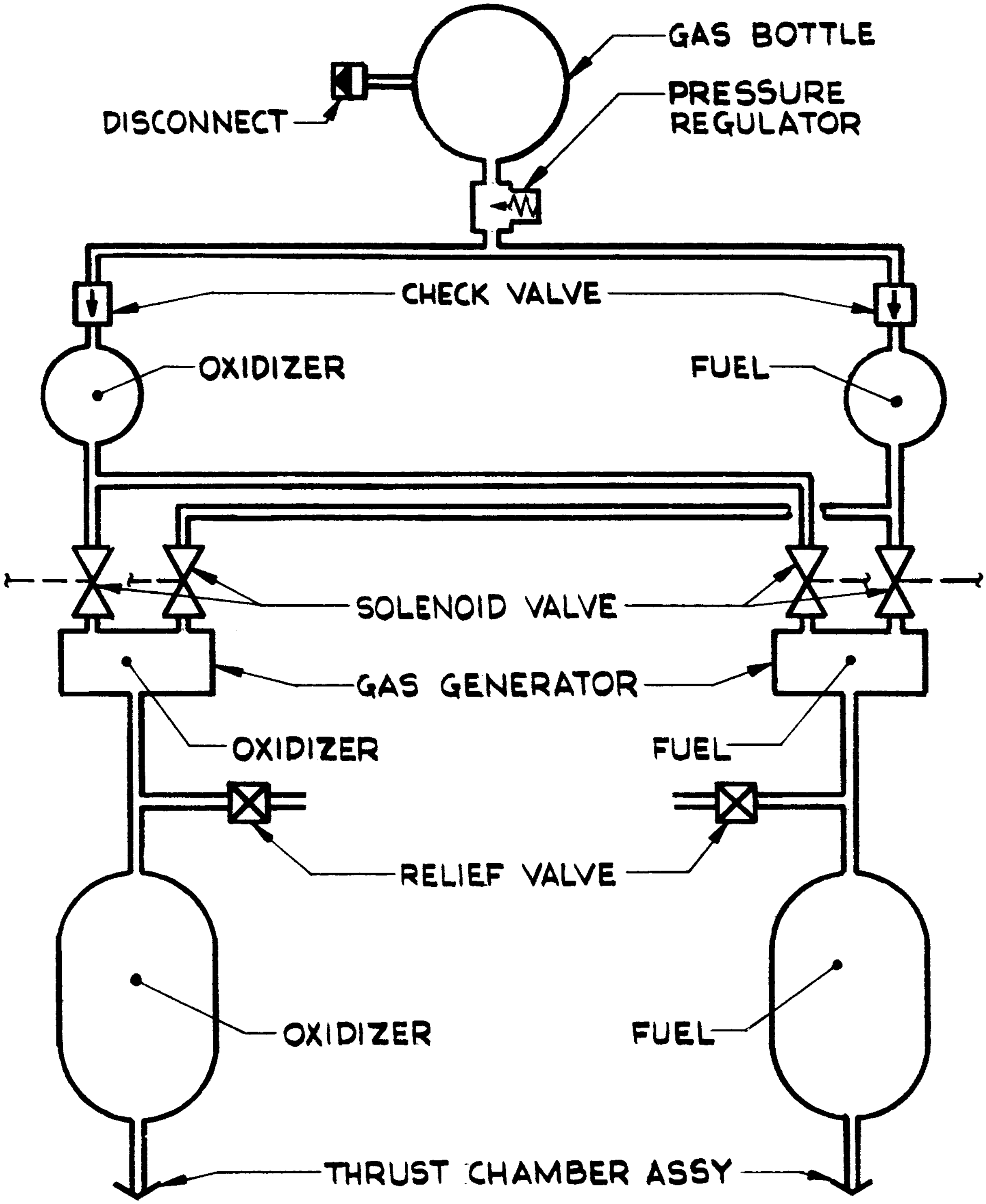
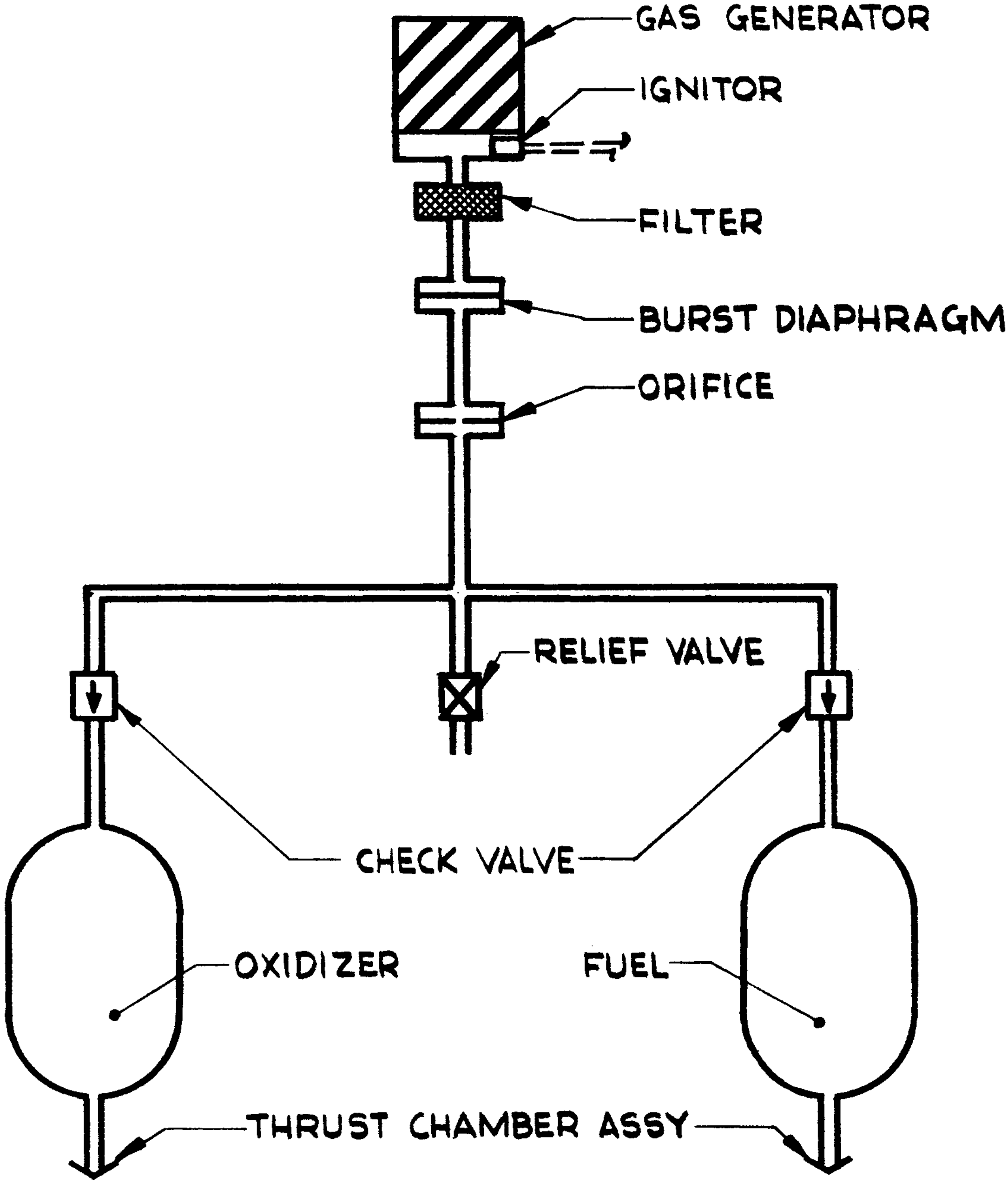


COMPONENT COMBINATION 10

Figure II-11



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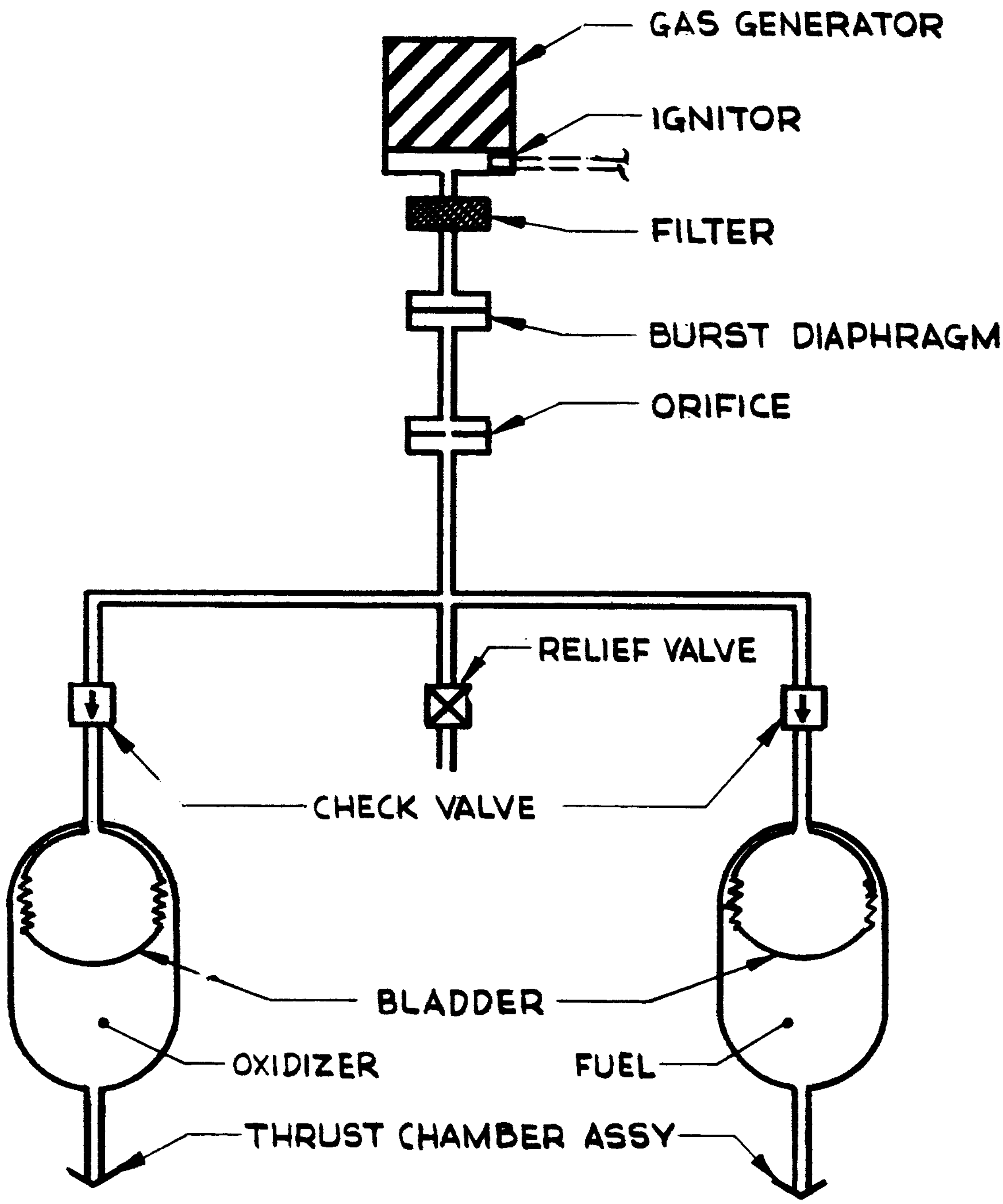

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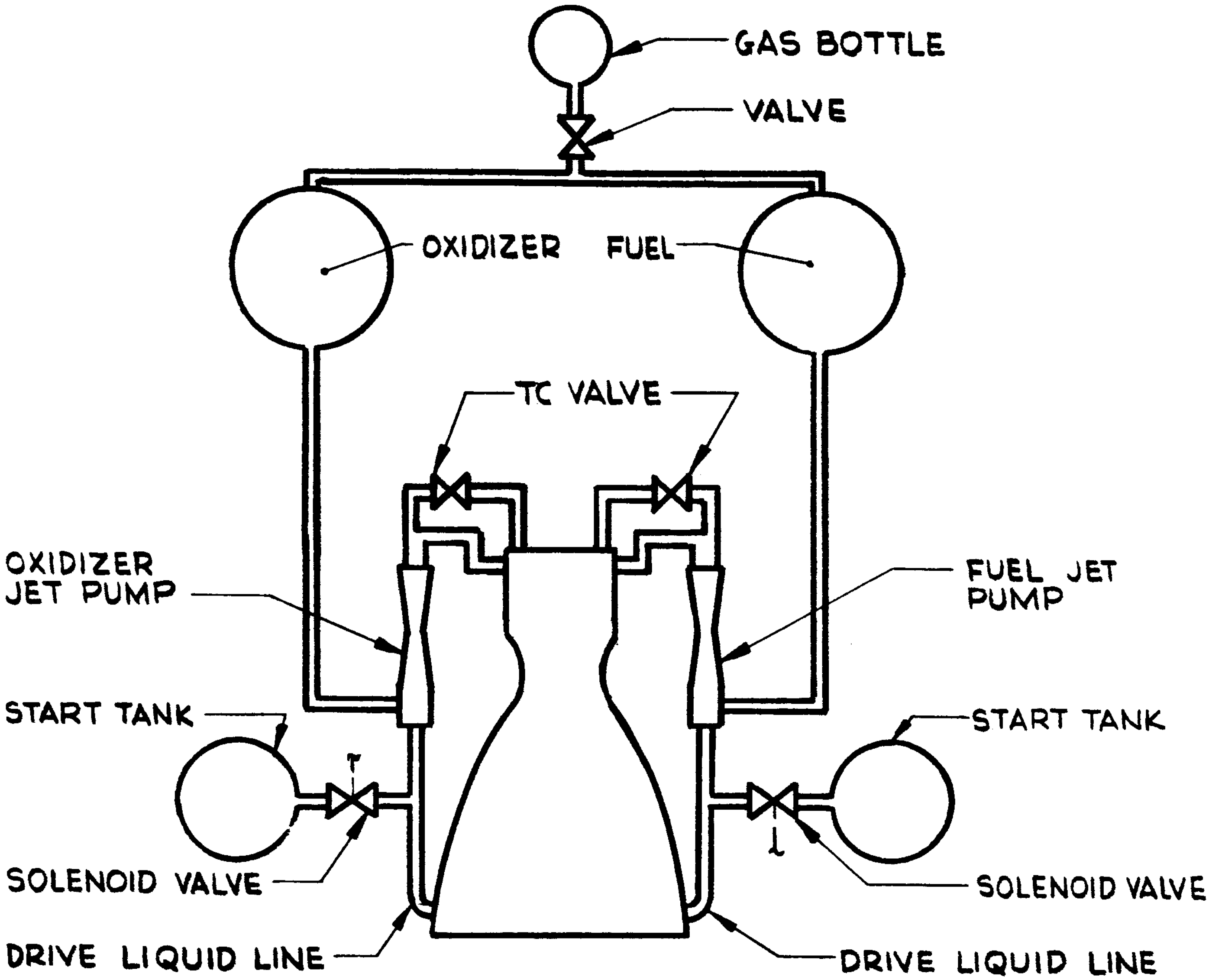

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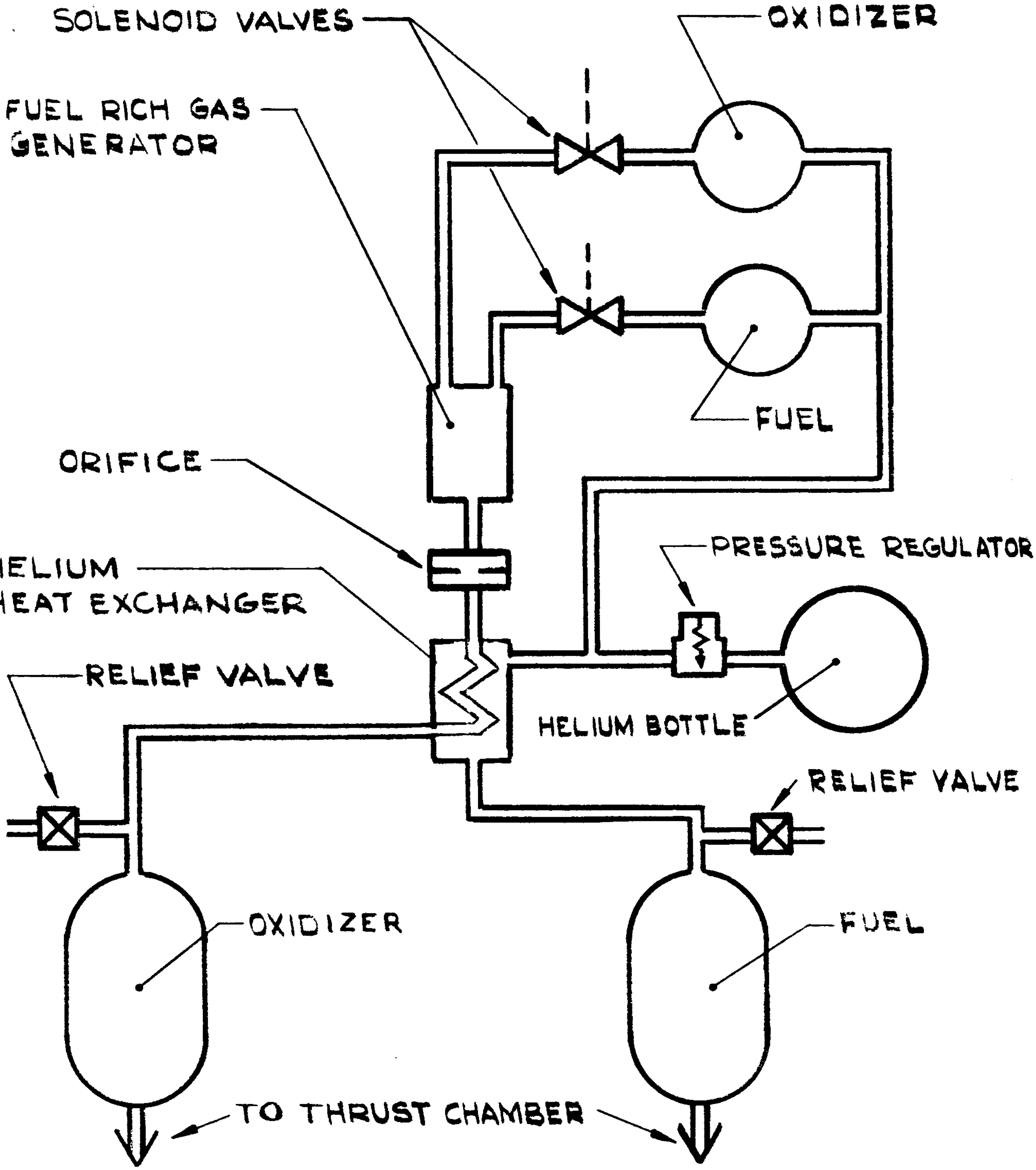

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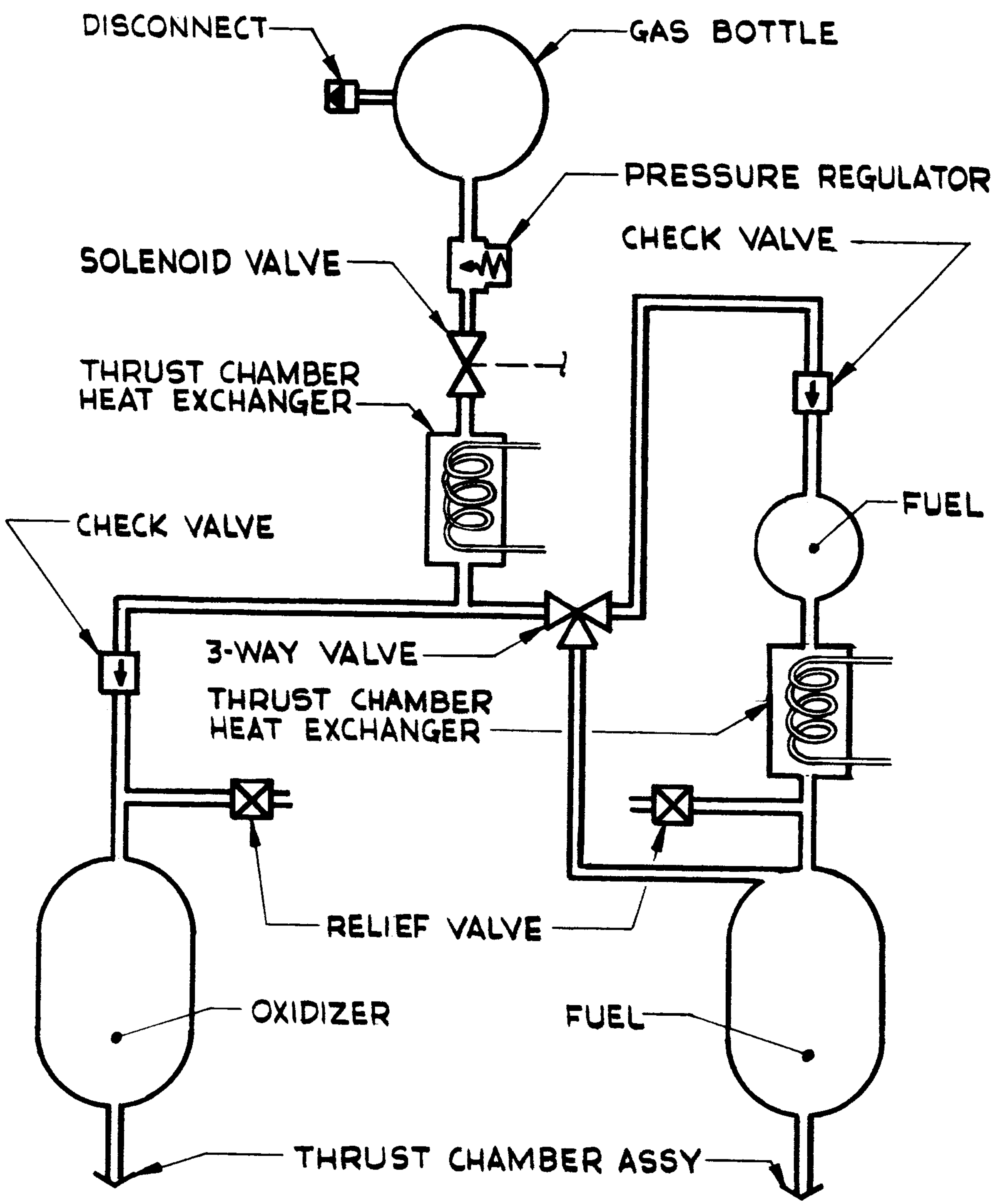
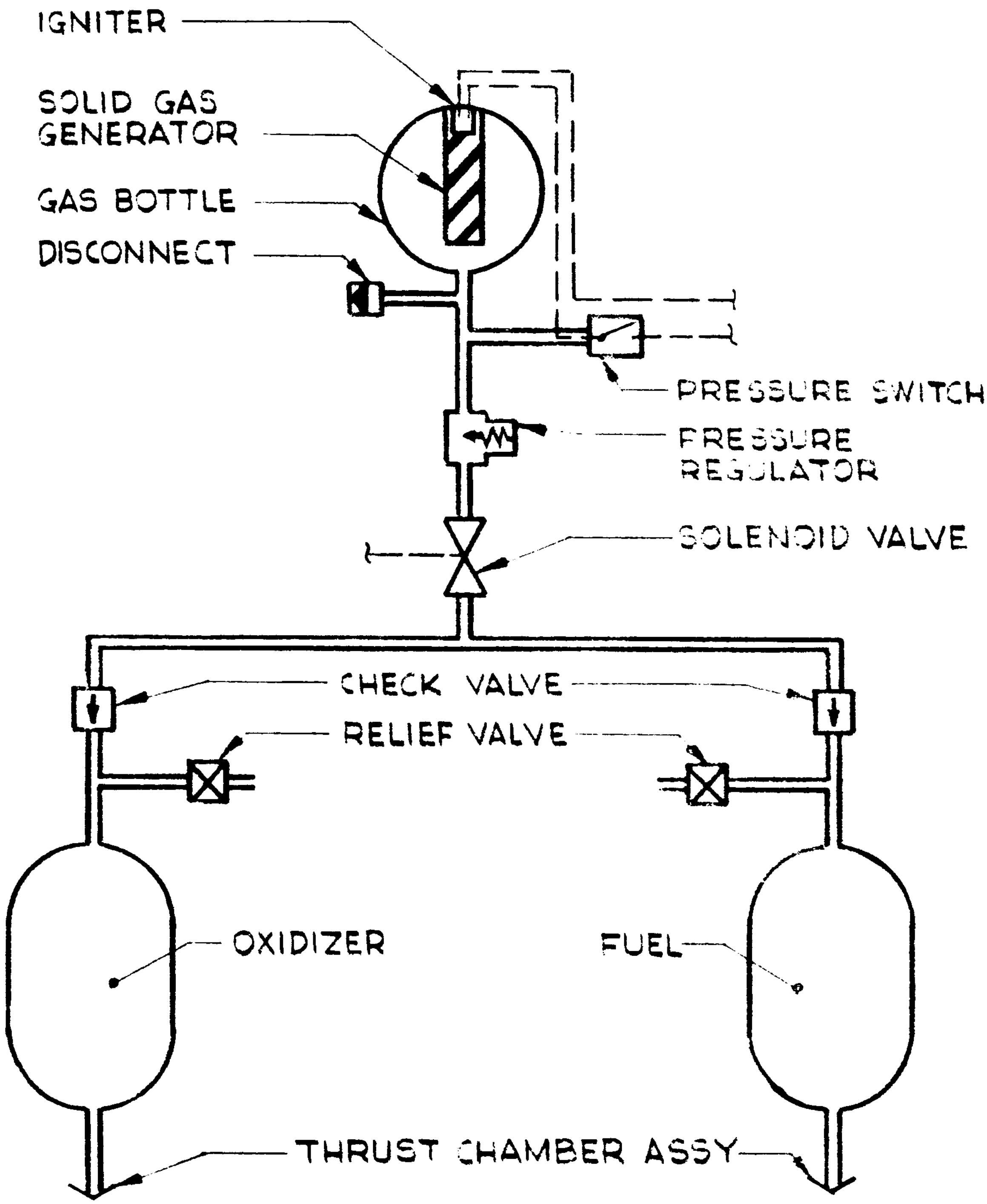


COMPONENT COMBINATION 15



COMPONENT COMBINATION 16

III. SYSTEM EVALUATION TECHNIQUE*

For any given mission, several propellant pressurization systems may be capable of meeting the performance requirements to a greater or lesser extent. The following rating technique has been devised to provide an objective means of comparing and selecting the most suitable pressurization systems for any mission.

A numerical rating, based upon performance factors, is determined for each candidate pressurization system. The final rating of each system is computed by multiplying a base value by the rating factors for that system. Two types of rating factors are established; qualitative factors, which systems must meet to be acceptable, and quantitive factors, which systems can fulfill to varying degrees. Examples of the two categories are shown below:

Qualitative Factors	Quantitative Factors
Restart capability	Reliability
Variable-thrust capability	Weight
Propellant compatibility	Size
200-day storability	Cost
	Control accuracy

Some rating factors can be both qualitative and quantitative depending upon the mission requirements. For example, if a minimum reliability of 97% were a requirement, all systems having reliabilities below this value would be eliminated from consideration; however, those systems with reliabilities above 97% would be rated quantitatively over the range of 97 to 100%.

This evaluation is maintained as an objective technique by establishing the rating factors (or influence coefficients) independently of and previous to the evaluation of system performance. Influence coefficient curves and tables are prepared to reflect the desired propellant pressurization system configuration, and the rating technique serves as a measure of how closely each candidate system approaches these desired values.

A. QUALITATIVE EVALUATION FACTORS

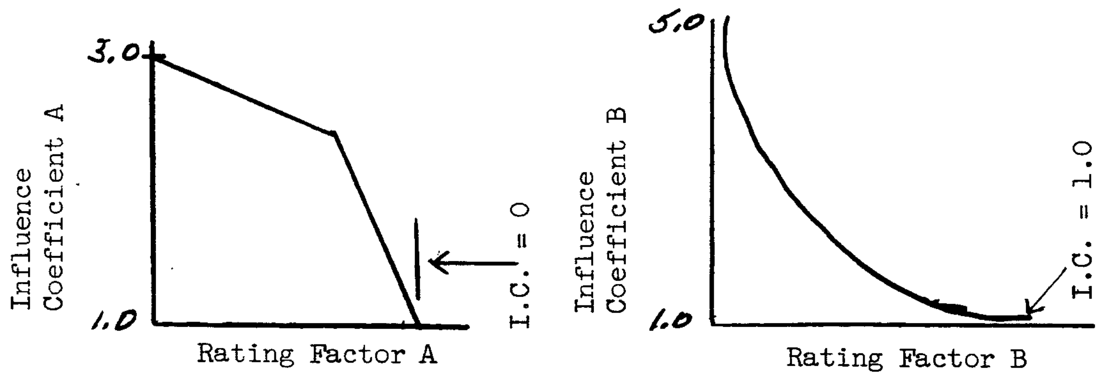
Qualitative factors are those rating parameters which are "go, no-go" measurements. If a system can meet a requirement it will rate 1.0, if not, it will

*This section will also appear in Aerojet Report No. 2334.

rate 0. Since the final numerical rating of the system is the product of the coefficients, a zero rating of any coefficient will eliminate that system from further consideration. The effect of this initial screening will be to reduce the number of candidate systems to a workable group.

B. QUANTITATIVE EVALUATION FACTORS

The remaining candidate systems are all capable of satisfying the mission requirements to varying extents. The quantitative evaluation factors will be presented as influence coefficient curves like those shown below:



The shape of the influence coefficient curves is a measure of the absolute importance that is placed upon an increase or decrease in the value of each rating factor. The rating factors may carry different weights in the overall evaluation; thus, the relative importance of each factor can be adjusted by varying the range of the influence coefficients on the ordinate of the curve. Rating factor A may have a range of influence coefficient from 1.0 to 3.0 while factor B may have a range of influence coefficient from 1.0 to 5.0, indicating that factor B has more influence on the selection of the system than factor A.

The value of the influence coefficient is defined as zero for rating factor values beyond the point where the value of the influence coefficient drops below 1.0. Thus, qualitative influence coefficient curves may be extended to represent both qualitative and quantitative considerations.

C. INFLUENCE COEFFICIENT USAGE

To illustrate the method of preparing the influence coefficient curves, a selection of a system for a manned, lunar mission will be demonstrated. Reliability, weight, and size will be the factors used in rating the systems.

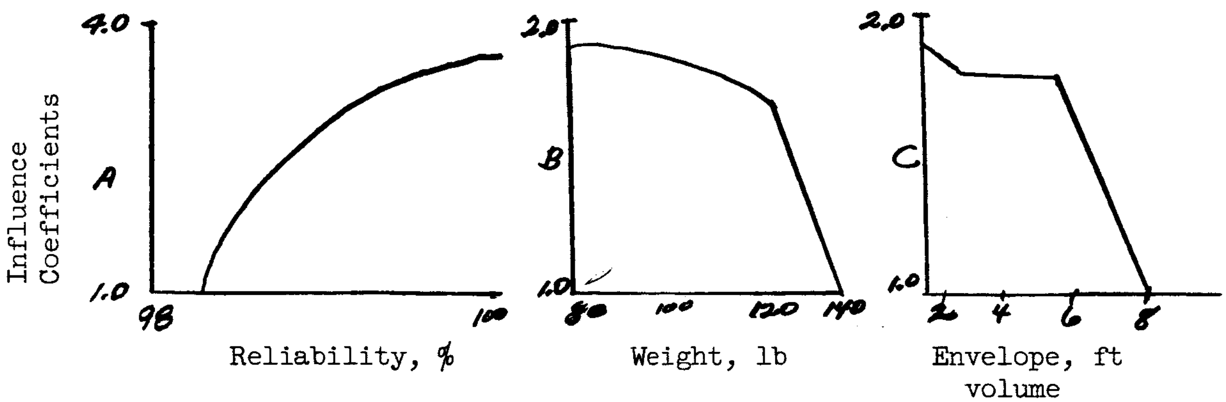
Minimum allowable reliability - 98.5%
Desired weight - 120 lb or less
Desired size - 6 ft^3 or less

1. Selection of Coefficient Ranges

Of the three rating factors, reliability is the most important for this manned mission, with weight and size being of lesser importance. The coefficient ranges are selected as follows:

Reliability	1.0 - 4.0
Weight	1.0 - 2.0
Size	1.0 - 2.0

2. Determination of Influence Curves



With the coordinates determined, the shape of the influence coefficient curves becomes a function of desired performance. A small improvement in reliability is highly desirable so the curve will exhibit a steep slope above the minimum value of 98.5%.

Variations in weight immediately above and below the desired value of 120 lb have a severe effect on the weight influence coefficient; however, a further decrease in weight below 100 lb is of little importance and the curve

levels out sharply at this point. A weight of 140 lb is the maximum which can be accepted and a quantitative cutoff is made in the curve.

An envelope of 6 ft^3 has been allotted to the system in one area of the vehicle. If it is larger than this, other equipment can be moved to provide a maximum of 8 ft^3. However, there is no advantage to a 4 ft^3 system since it would still occupy the same location. Below 4 ft^3, the system can be installed in several unused areas and there is an advantage to small-size systems. The volume curve slopes sharply from 8 ft^3 down to 6 ft^3; then, it is flat from 6 to 4 ft^3 and slopes sharply again below 4 ft^3.

3. Final Evaluation and System Selection

With the rating curves prepared, the reliabilities, weights, sizes, etc. of each system are determined using the data presented in Volume III, Report No. 2334. These values are applied to the influence coefficient charts and the resulting coefficients are tabulated as shown below.

	Influence Coefficients				
	Base	A	B	C	Point Rating
System 1	10	3.2	1.7	1.6	87
System 2	10	1.8	1.8	1.2	39
System 3	10	3.0	1.1	1.6	53
System 4	10	2.4	1.8	1.8	78

The numerical rating of each candidate system is determined by multiplying a base value of 10.0 by the product of the influence coefficients. The system with the highest point rating is the one most suitable for the mission.

In the sample case, System 1 with a point rating of 87 would be the best system to accomplish the mission. Viewing the tabulation reveals the strong and weak points of each system. It should be noted that System 1 rated highest only under Factor A; however, Factor A was of high final rating. This might be typical of the reliability factor on a man-rated vehicle.

Use of the influence coefficient method for evaluating systems, organizes the thought behind system selection and removes evaluation from the realm of intuition. The influence coefficient curves permit a review and discussion of the factors attendent to the final selection without considering a particular pressurization system. The curves, themselves, are the result of a subjective definition of the mission which, once established, provide a valuable tool for the objective selection of the most suitable system.

IV. SYSTEM ANALYSIS

A. MISSION REQUIREMENTS

Propulsion system requirements are based on the results of Contract NAS 5-915 (mission analysis study). The mission analysis determines the thrust and total impulse range considered in this study. The requirements are tabulated in Table IV-1. These requirements, in conjunction with propellant performance data for the propellant combinations selected, are used to derive the pressurization system analysis parameters. Following is an outline of the derivation of some of the system parameters.

Volumetric propellant flow rate is a parameter required to size and determine the operating characteristics of all flow components such as regulators, heat exchangers, tubing, and valves. The volumetric flow rate is determined in the following manner:

$$Q = F/I_{sp} \times \text{bulk density}$$

Q = volumetric flow rate, cfs

I_{sp} = specific impulse, lb_f/lbm/sec

F = thrust, lb_f

bulk density = density of propellant combination, lbm/ft^3

The total volume of the propellant tank, together with the tank pressure and pressurant "average" temperature determine the quantity of pressurant required for a particular mission. This, in turn, determines the size of the storage container or gas generator. The total propellant tank volume is calculated by multiplying the volumetric flow rate by the duration of firing.

$$t = I_t/F$$

$$V_t = Q \times t$$

t = duration of firing, sec

I_t = total impulse, lb_f-sec

V_t = total propellant tank volume, ft^3

The system performance limits were established only for the unmanned missions studied. These limits are as follows:

1. The minimum volumetric flow rate and tank volume are required for the lunar terminal correction maneuver. Using ClF_3/Hydrazoid as propellants, the limits are

$$Q = 0.02 \text{ cfs}; \quad V_t = 1.26 \text{ ft}^3$$

2. The maximum volumetric flow rate and tank volume are required for the Mars takeoff maneuver. Using LO_2/LH_2 as propellants,

$$Q = 7.5 \text{ cfs}; \quad V_t = 755 \text{ ft}^3$$

B. PROPELLANT SELECTION AND PERFORMANCE DATA

The mission analysis study determined the thrust and total impulse range to be considered in this study. The range of thrust varies from 25 to 66,000 lb. Calculations were made to determine the fuel and oxidizer flow rates for this range of thrust. From the range of flow rates, tank weights and volumes may be determined.

The calculations are based on equations, data from curves, and assumptions as outlined below.

1. Equations

From the thrust equation

$$F = \dot{W}_t \left[\frac{v_e}{g} + \Delta P \left(\frac{A_e}{A_t}\right) \frac{C^*}{Pcg} \right]$$

where

F = thrust, lb

$\dot{W}_t$ = total propellant flow, lb/sec

v_e = velocity at exit plane of exhaust nozzle, ft/sec

g = constant = 32.17 ft/sec^2

ΔP = exhaust nozzle exit plane pressure, ambient pressure (Pe-Pa)

Ae/At = exhaust nozzle area ratio, exit area/throat area

C^* = characteristic velocity, ft/sec

Pc = chamber pressure, psia

$\dot{W}_o$ = oxidizer flow, lb/sec

$\dot{W}_f$ = fuel flow, lb/sec

P_e = exhaust nozzle exit plane static pressure, psia

P_a = ambient pressure, psia

2. Assumptions

a. Equilibrium expansion in the nozzle

b. Exhaust nozzle exit plane pressure (P_e) = 1.0 psia

c. Ambient pressure (P_a) = zero

From the above assumptions, and for a given propellant combination, $\dot{W}_t$ is a straight line function of thrust. Therefore, values were calculated for the maximum thrust and a curve of F vs $\dot{W}_t$ was constructed. Values of exhaust nozzle-to-chamber pressure ratios of 100 to 500 in increments of 100, were considered for four propellant combinations.

It is felt that four propellant combinations should be included in this study. Three of these are the more advanced storables and one is a cryogenic. These combinations are listed below:

LO_2/LH_2

N_2O_4/Aerozine-50

N_2O_4/B_5H_9

ClF_3/Hydrazoid

3. Data From Curves*

From the curve of v_e vs $\dot{W}_o/\dot{W}_f$ with P_c/P_e parameters, values were obtained for the maximum v_e. Corresponding values of C^* and A_e/A_t were obtained from the other curves plotted in a previous Aerojet report.*

*Performance and Properties of Liquid Propellants, Aerojet-General Corporation, Report No. 8160-6S, 6 March 1961.

Specific gravities of the propellants were chosen* and are tabulated in Table IV-2. Also tabulated, and presented in Table IV-3, are the results of the calculations performed at the stated pressure ratios and propellant combinations.

The following curves were plotted from the above calculations:

a. Figure IV-1, propellant mass ratio vs exhaust nozzle pressure ratio with propellant combination parameters

b. Figure IV-2, exhaust nozzle area ratio vs exhaust nozzle pressure ratio with propellant combination parameters

c. Figure IV-3, specific impulse vs exhaust nozzle pressure ratio with propellant combination parameters.

C. PRESSURIZATION SYSTEM OPERATING CHARACTERISTICS AND DESIGN CRITERIA

1. Design Criteria

The minimum number of design criteria required to describe and select a pressurization system have been determined. These criteria can be grouped into three categories:

a. Operational

(1) Restart

One of the evaluation parameters that will influence the selection of a pressurization system for most of the space missions to be considered is restart capability. The restart capability of each pressurization system is listed in Table IV-4.

The only system that is non-restartable is the solid-propellant, gas-generation system, and the solid-propellant, gas-generation system with bladder. It is conceivable that this system could be made restartable by the addition of another gas-generator unit; however, in so doing it is felt that the system will not be competitive with the other systems due to the increased weight and volume and the decrease in reliability. Presently, concepts

*ibid.

of stopping the burning of the solid-propellant grain in order to make the system restartable have been conceived. One of the proposed methods would be to de-pressurize the gas generator to a level where the solid-propellant grain would cease to burn. Ignition for restarting the gas generator could be accomplished by injecting liquid propellant upon the burning grain surface.

(2) Storability - before ignition and between restarts

(3) Space environment - temperature, pressure and duration

(4) Zero "G"

b. Performance

(1) Propellants - mixture ratio, I_{sp}, and density

(2) Thrust

(3) Combustion-chamber pressure

c. Materials

The parameters used in the comparative evaluation of the propellant pressurization systems, i.e., weight, volume, and propellant compatibility, dictate the evaluation of feasible materials. The results of the material evaluation will enable the design of a system for a given mission and propellant combination.

A wide range of data is required because of the environmental extremes that will be encountered in the study of pressurization systems with many propellants and varied missions.

Tank and associated equipment sizes and weights are dictated by the strength of the material and the operating pressures. Strength varies with environmental temperature, and the temperatures considered in this study will vary from -423 to +2500°F. Therefore, depending on the surrounding temperature, one material will usually stand out as the one to be considered for use.

To determine the temperature of the bulk gas after expansion in the propellant tanks, the specific heat of the tankage material must be known in order to solve the heat-balance equation. The final bulk-gas temperature will be a factor in making the material choice for a given mission.

Corrosion data must be factored into the design of a pressurization system because of long storage times encountered in some missions. Some materials are not compatible with some propellants; therefore these data must be made available. However, it is usually a yes or no decision except in short-duration applications when some degree of incompatibility can be tolerated.

Using the information above as input, the design handbook will enable the user to select the best candidate system and then complete a preliminary design of that system. A computer program could use this same input to perform the selection and preliminary design functions.

2. Pressurization System Operating Characteristics

a. Summary

Operating characteristics include evaluation of the following factors:

(1) Thermodynamic flow processes

(2) Pressurant requirements

(3) Size and weight of components

(4) Operating temperatures and pressures

(5) Heat exchanger and regulator performance.

Whenever possible, operating characteristics are described as a function of volumetric gas flow and pressurization work. Both of these quantities are derived directly from mission study data. Volumetric flow is a measure of the required flow area of all operating system components and is therefore a function of their size and weight. Its derivation is shown in paragraph D,1 following. Pressurization work, propellant-tank volume times pressure, is a measure of the energy required to expel the propellant. This term provides a means of comparing the various types of pressurization systems; in stored gas systems it is a measure of the quantity of high-pressure gas, in the gas-generator systems it is a measure of the required propellant, and in the positive displacement systems it is a measure of the mechanical energy needed.

D. EXAMPLE, STORED-GAS SYSTEMS

The flow chart, Figure IV-4, shows the steps that were used in determining the weight of stored-gas systems. Starting with input from the mission study, propellant-tank pressure and volume and volumetric propellant flow rate can be determined. All component weights will be determined as a function of these parameters.

1. Volumetric Flow Rate

Given the thrust level and propellant combination, the volumetric propellant flow rate is determined by

$$\text{Flow rate} = \text{thrust}/I_{sp} \times 1/\text{bulk density}$$

This relationship is shown graphically in Figure IV-5 for the propellant combinations being considered in this study.

2. Pressure Loss, Tank to Chamber

The combustion-chamber pressure, an engine parameter, is related to propellant-tank pressure as shown in Figure IV-6. Test data from existing missiles were applied to pressure-drop equations to develop this curve.

3. Propellant-Tank Volume

Multiplying the propellant volumetric flow rate by the firing duration (Q x t) completes the determination of the comparison parameters for any mission.

4. Pressurization-Gas Requirements

Using the pressurization work, tank pressure times volume, Figures IV-7, IV-8, and IV-9 can be used to determine the weight of pressurizing gas for each stored-gas system. Each system has an optimum operating temperature, and it is this temperature which would be used on the curves. As tank pressures are not expected to exceed 1000 psia, compressibility effects are not included and the curves represent the equation

$$M = PV/RT$$

5. Residual Gas Requirements

In a stored-gas system some portion of the gas supply is "unavailable" for use in pressurizing the propellant tanks because of its low pressure. This unavailable or residual gas may comprise as much as 25% of the total.

The ratio of initial to final mass after expansion from a high-pressure, stored-gas source depends upon some expansion coefficient or path to define the end points.

$$\text{Initial mass} = \frac{P_1 \ V_1}{Z_1 \ RT_1} = m_1$$

$$\text{Residual mass} = \frac{P_2 \ V_2}{Z_2 \ RT_2} = m_2$$

$$m_2/m_1 = \frac{P_2 \ V_2}{Z_2 \ RT_2} \times \frac{Z_1 \ RT_1}{P_1 \ \ V_1} = Z_1 \ \frac{P_2}{P_2} \ \frac{T_1}{T_2}$$

For final pressures considered here, $Z_2 = 1.0$

With P_1, T_1 and P_2 specified, T_2 may be determined from a polytropic efficiency definition as follows:

$$\eta \text{ poly} = \frac{c_p \ln (T_1/T_2)}{R \ln (P_1/P_2)} = \frac{\gamma}{\gamma - 1} \ \frac{\ln T_1/T_2}{\ln P_1/P_2}$$

$$\frac{T_1}{T_2} = \frac{P_1}{P_2} \exp \left[\frac{\gamma - 1}{\gamma} \ \eta \text{ poly} \right]$$

$$m_2/m_1 = Z_1 \ (P_2/P_1) \ \exp \left[\frac{1 - \gamma}{\gamma} \ \eta \text{ poly} + 1 \right]$$

The range of the polytropic coefficient is from 0 to 1. The actual value must be determined experimentally. Usual values for flight η poly are from 0.4 to 0.6 with flight-weight hardware and stored-gas systems.

As an example, assuming helium η poly = 0.572 for a typical stored-helium system

$$\text{Helium} \quad \frac{\gamma - 1}{\gamma} = 0.399$$

$$T_1 = 520^{\circ}F$$

$$P_1 = 4500 \text{ psia}$$

$$Z_1 = 1.153$$

P_2 - psia	T_2 - $^{\circ}R$	T_1/T_2	m_2/m_1
800	350	1.484	0.294
600	328	1.583	0.237
400	300	1.738	0.195
200	256	2.034	0.103
100	218	2.380	0.061

Residual gas weights for helium, hydrogen, and nitrogen are shown in Figure IV-10.

6. Bottle Weight

The weight of the high-pressure spherical "bottle" can be derived as a function of the weight of gas stored and the container material.

Bottle weight has been based on the outside diameter rather than on the median diameter of the sphere. The resultant weight is a very close approximation of bottle weight with pressure ports added.

Bottle weight = surface area x wall thickness x material density

$$A = 4\pi r^2; \quad t = Pr/2S; \quad r = (3V/4\pi)^{1/3}$$

Substituting

$$Wt = 4\pi r^2 \times (Pr/2S)\ \rho = 4\ \pi\ r^3\ (P/2S)\ \rho$$

$$= 4\pi\ (3V/4\pi) \times (P/2S)\ \rho$$

$$= 3/2 \times P\ \rho/S \times MZRT/P$$

Bottle weight = 18 MZRT ρ/S x SF

M = mass of stored gas, lb_m

R = gas constant of stored gas, ft-lb_f/lb_m - ^oR

T = stored gas temperature, ^oR

Z = compressibility factor

ρ/S = density to strength ratio of bottle material, lb_m/in.-lb_f

SF = safety factor on yield strength

7. Component Weight

The weights of available and future pressurization system components are being compiled for inclusion in this study. The following method has been developed for scaling these weights for various size systems.

Typical units are selected and their weight, flow capacity, pressure drop, and operating pressure are recorded. These units are then scaled to the operating pressures and flow capacities of interest in this study. To determine the effect of pressure on weight, the components are treated as cylinders. With a constant pressure drop maintained, the effect of flow on weight was determined for the components. The weight variation of the pressure regulator is shown in Figure IV-11.

8. Scaling Factor

a. Effect of change in flow rate and pressure with constant temperature and constant length.

Subscript o = typical unit conditions

Subscript s = scaled-up (or down) conditions

$$Wto/Wts = \frac{\rho_o V_o}{\rho_s V_s} = \frac{D_o t_o L_o \rho_o}{D_s t_s L_s \rho_s} ; \; t = \frac{PD}{2s}$$

$$= \frac{D_o^2 \; L \; P_o \; 2S_\rho}{D_s^2 \; L \; P_s \; 2S_\rho} = (D_o/D_s)^2 \; P_o/P_s$$

$$Wto/Wts = (D_o/D_s)^2 \; P_o/P_s$$

b. To maintain constant pressure drop,

$$\frac{\rho_o v_o^2}{2g} \; f_o L_o/D_o = \frac{\rho_s v_s^2}{2g} \; f_s L_s/D_s ; \quad v = W/\rho A = Q/A$$

but L, f and ρ are constant.

$$Q_o^2/A_o^2 D_o = Q_s^2/A_s^2 D_s ; \; A = \pi D^2/4$$

$$Q_o^2/D_o^5 = Q_s^2/D_s^5$$

$$(Q_o/Q_s)^2 = (D_o/D_s)^5$$

$$(D_o/D_s)^2 = (Q_o/Q_s)^{4/5}$$

Then, from the weight equation of the previous section

$$Wt_o/Wt_s = (Q_o/Q_s)^{4/5} \; (P_o/P_s)$$

$$Wt_s/ = Wt_o \; (Q_s/Q_o)^{4/5} (P_s/P_o)$$

Equation for scaling weight to account for changes in pressure, gas medium, and volume flow rate

$$Wt_o/Wt_s = (P_o/P_s) \; (\rho_o/\rho_s)^{2/5} \; (Q_o/Q_s)^{4/5}$$

$$Wt_s = Wt_o \; (P_s/P_o) \; (\rho_s/\rho_o)^{2/5} \; (Q_s/Q_o)^{4/5}$$

c. Utilization Factor

(1) The pressurizing gas is typically used at temperatures above that of the propellants in the tank. An accurate assessment of the weight of pressurant required can only be made if the cooling effect of the propellant tank upon the pressurizing gas is known. The derivation of the utilization factor, which is the ratio of the inlet-gas temperature to the final bulk-gas temperature, is shown in the appendix. The basic assumptions made in this analysis are brought forth in the derivation.

(2) Using the derived relationship for the pressurization utilization factor, the inlet-gas temperature vs the final bulk-gas temperature was plotted for tank materials - titanium 6Al4V and stainless steel 17-7 PH. The pressurant gases used were nitrogen, hydrogen, and helium. These curves are presented in Figures IV-12 and IV-13.

(3) The assumptions made in the calculation of the utilization factor are as follows:

(a) Storable propellants used

(b) Propellant-tank temperature, initially 520°R

(c) Tank pressure, 300 psia

(d) Safety factor of propellant tank, 1.4 (x yield strength)

(e) Volume of propellant tank, 625 cu ft

(f) Density of Ti 6Al-4V, 0.160 lb/in.^{3}

(g) Density of SS 17-7 PH, 0.276 lb/in.^{3}

(h) Initial ullage volume of propellant tank, 1%

(4) The curves depicting the specific heat ratios of N_2, H_2 and He as a function of temperature and pressure are shown in Figures IV-14, IV-15 and IV-16, respectively.

(a) The specific heat of Ti 6Al-4V and SS 17-7 PH as a function of temperature is shown in Figure IV-17.

(b) The yield strength of Ti 6Al-4V and SS 17-7 PH vs temperature is presented in Figure IV-18.

TABLE IV-1

RESULTS OF MISSION ANALYSIS

Maneuver	Thrust (lb_F)	Total Impulse (lb_F-sec)
A. Orbital Correction		
1. Orbital Perturbations		
a. Atmospheric drag		
b. Earth oblateness effects	1000	0.24×10^6
2. Eccentricity control	4000	0.24×10^6
3. Orbital plane change	4000	0.24×10^6
4. Orbital altitude variation	4000	0.24×10^6
5. Orbital epoch change	4000	0.24×10^6
6. Correction of injection errors	2000	0.12×10^6
B. Orbital Rendezvous		
1. Nominal injection errors	4250	0.16×10^6
2. Dog-leg-"maneuver"	25,500	1.3×10^6
3. Emergency rendezvous	14,000	1×10^6
C. Trajectory Corrections		
1. Midcourse corrections		
a. Lunar flights	125	7750
b. Planetary flights (Mars - Venus)	500	0.15×10^6
c. Planetary return flights (Mars - Venus)	150	45,600
2. Terminal corrections		
a. Lunar flights	125	7750
b. Planetary flights (Mars)	500	31,000
(Moon)	25	1550
c. Return flights (Mars)	150	9300
D. Orbiting Maneuvers		
1. Moon orbits (no atmos)	2500	0.26×10^6
2. Mars orbits (with atmos)	18,000	1.6×10^6

TABLE IV-1 (cont.)

Maneuver	Thrust (lb_F)	Total Impulse (lb_F-sec)
E. Landings		
1. Lunar landings		
a. Direct	2300	0.45×10^6
b. From orbit	1500	0.21×10^6
2. Mars landing		
a. Direct	18,000	2.2×10^6
b. From orbit (with atmos)	14,000	0.79×10^6
F. Takeoffs		
1. Lunar takeoffs		
a. To orbit	3000	0.42×10^6
b. Direct to earth	4500	0.59×10^6
2. Mars takeoffs		
a. To orbit	15,000	2.4×10^6
b. Direct to earth	66,000	6.5×10^6 1st stage
	22,000	2.6×10^6 2nd stage

Table IV-1
Sheet 2 of 2

TABLE IV-2

SPECIFIC GRAVITY OF PROPELLANTS

Fuels	Specific Gravity at °F
Aerozine-50	0.905 at 60°F
Hydrazoid	1.092 at 60°F
LH_2	0.073 at -423°F
B_5H_9	0.633 at 60°F
Oxidizers	
N_2H_4	1.45 at 60°F
ClF_3	1.85 at 60°F
LO_2	1.14 at -297°F

TABLE [illegible]-3

PROPELLA[illegible]

PERFORMANCE CA[illegible]LATIONS

F = 66,0[illegible] lbs

Propellant Combination	LO_2 / LH_2	N_2O_4 / A[illegible]zine - 50	N_2O_4 / B_5H_9	ClF_3 / Hydrazoid
Pc/Pe - dim	500	500	500	500
v_e - ft/sec	14,080	[illegible]180	11130	10180
Wo/Wf - dim	4.6	[illegible].13	3.33	2.4
C* - ft/sec	7870	[illegible]540	5820	5781
Ae/At - dim	36.4	4.4	54.4	32.4
Wt - lbs/sec	145.0	[illegible]3.3	180.6	201.0
Wf - lbs/sec	25.9	[illegible]1.5	41.7	59.1
Wo - lbs/sec	119.1	[illegible]1.8	138.9	141.9
Isp - sec.	455	[illegible]341	365	328

Propellant Combination	LO_2 / LH_2	N_2O_4 / [illegible]rozine - 50	N_2O_4 / B_5H_9	ClF_3 / Hydrazoid
Pc/Pe - dim	400	400	400	400
v_e - ft/sec	13,950	[illegible],380	10970	10,090
Wo/Wf - dim	4.6	2.16	3.36	2.38
C* - ft/sec	7870	5612	5800	5765
Ae/At - dim	31.2	35.4	45.1	27.7
Wt - lbs/sec	145.9	95.3	182.3	202.5
Wf - lbs/sec	26.0	61.8	41.85	59.9
Wo - lbs/sec	119.1	[illegible]33.5	140.45	142.60
Isp - sec.	452	338	362	326

Propellant Combination	LO_2 / LH_2	N_2O_4 / [illegible]rozine - 50	N_2O_4 / B_5H_9	ClF_3 / Hydrazoid
Pc/Pe - dim	300	300	300	300
v_e - ft/sec	13,750	[illegible],220	10,660	9950
Wo/Wf - dim	4.8	2.08	3.26	2.34
C* - ft/sec	7815	5628	5792	5758
Ae/At - dim	25.7	29.2	35.7	22.5
Wt - lbs/sec	147.1	197.3	187.0	204.5
Wf - lbs/sec	25.35	54.10	43.85	61.2
Wo - lbs/sec	121.75	133.2	143.15	143.3
Isp - sec.	449	334	353	323

TABI IV-3 (cont.)

PR LLANT

PERFORMA CALCULATIONS

F 6,000 lbs

Propellant Combination	LO_2 / LH_2	N_2O_4 / Aerozine - 50	N_2O_4 / B_5H_9	ClF_3 / Hydrazoid
Pc/Pe - dim	200	200	200	200
v_e - ft/sec	13,480	10,000	10,470	9780
Wo/Wf - dim	4.35	2.08	3.20	2.36
C* - ft/sec	7672	5616	5775	5735
Ae/At - dim	18.2	21.6	25.5	17.0
Wt - lbs/sec	149.2	200.0	189.4	206.6
Wf - lbs/sec	28.15	64.9	45.1	61.5
Wo - lbs/sec	121.55	135.1	144.3	145.1
Isp - sec.	441	330	348	319

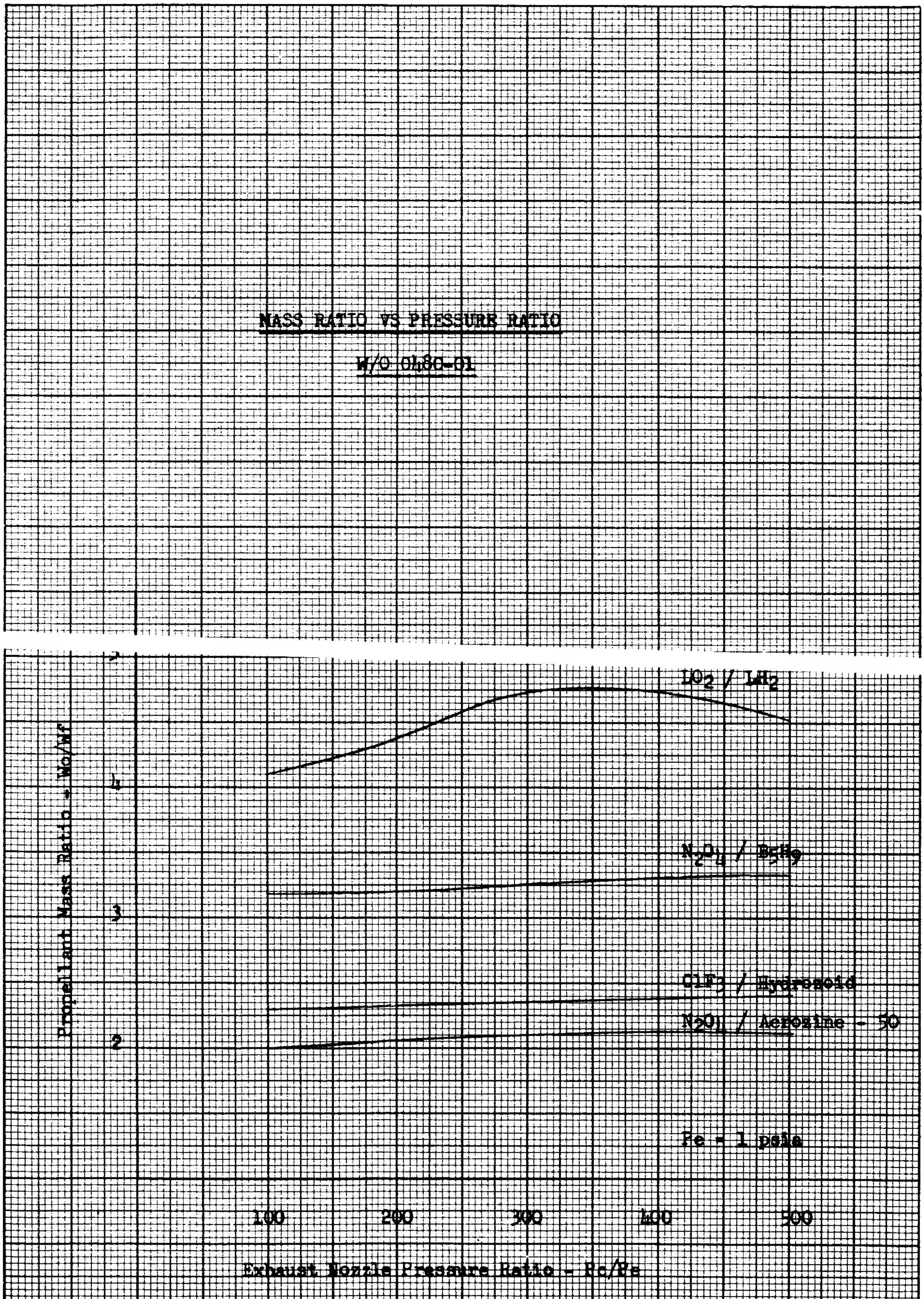
Propellant Combination	LO_2 / LH_2	N_2O_4 / Aerozine - 50	N_2O_4 / B_5H_9	ClF_3 / Hydrazoid
Pc/Pe - dim	100	100	100	100
v_e - ft/sec	12,950	9510	9930	9400
Wo/Wf - dim	4.1	2.02	3.18	2.34
C* - ft/sec	7670	5623	5762	5720
Ae/At - dim	10.7	12.5	14.4	10.6
Wt - lbs/sec	154.2	208.0	197.3	212.3
Wf - lbs/sec	30.2	68.9	47.15	63.6
Wo - lbs/sec	124.0	139.1	150.15	148.7
Isp - sec.	429	318	335	311

TABLE IV-4

RESTART CAPABILITY

SELECTED PRESSURIZATION SYSTEMS

System	Restart Capability	
Stored cold gas	Yes	
Heated gas	Yes	
Liquid-propellant gas generation	Yes	
Solid-propellant gas generation		No
Vaporization	Yes	
Propellant injection	Yes	
Line-heated gas	Yes	
Stored cold gas (with bladder)	Yes	
Heated gas (with bladder)	Yes	
Liquid-propellant gas generator (with bladder)	Yes	
Solid-propellant gas generator (with bladder)		No
Mechanical displacement	Yes	
Regenerative jet pump	Yes	


MASS RATIO VS PRESSURE RATIO
W/O 0480-01
LO2 / LH2
N2O4 / B5H9
ClF3 / Hydrazoid
N2O4 / Aerozine - 50
Pe = 1 psia
Propellant Mass Ratio - Wo/Wf
2
3
4
100
200
300
400
500
Exhaust Nozzle Pressure Ratio - Pc/Pe

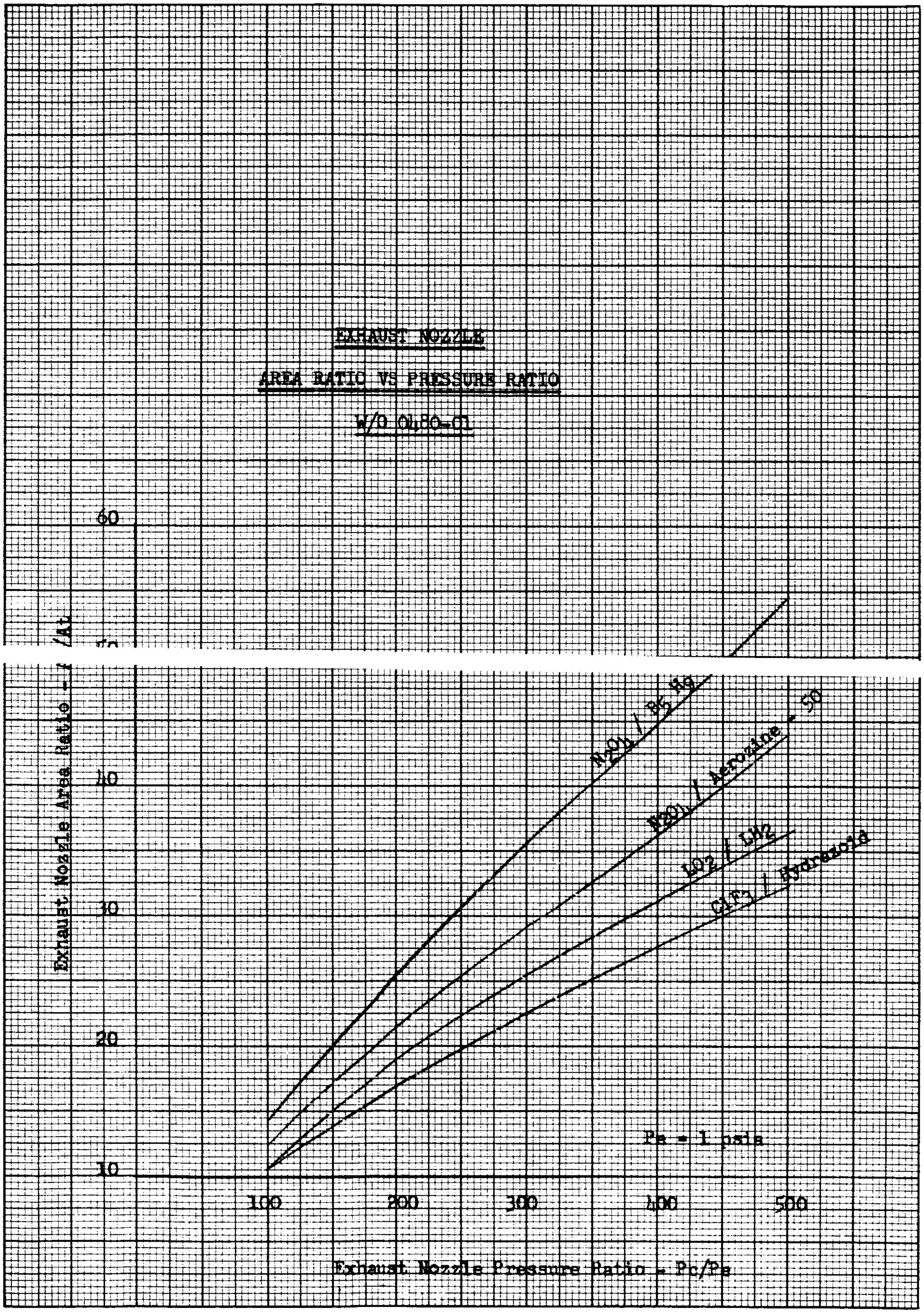
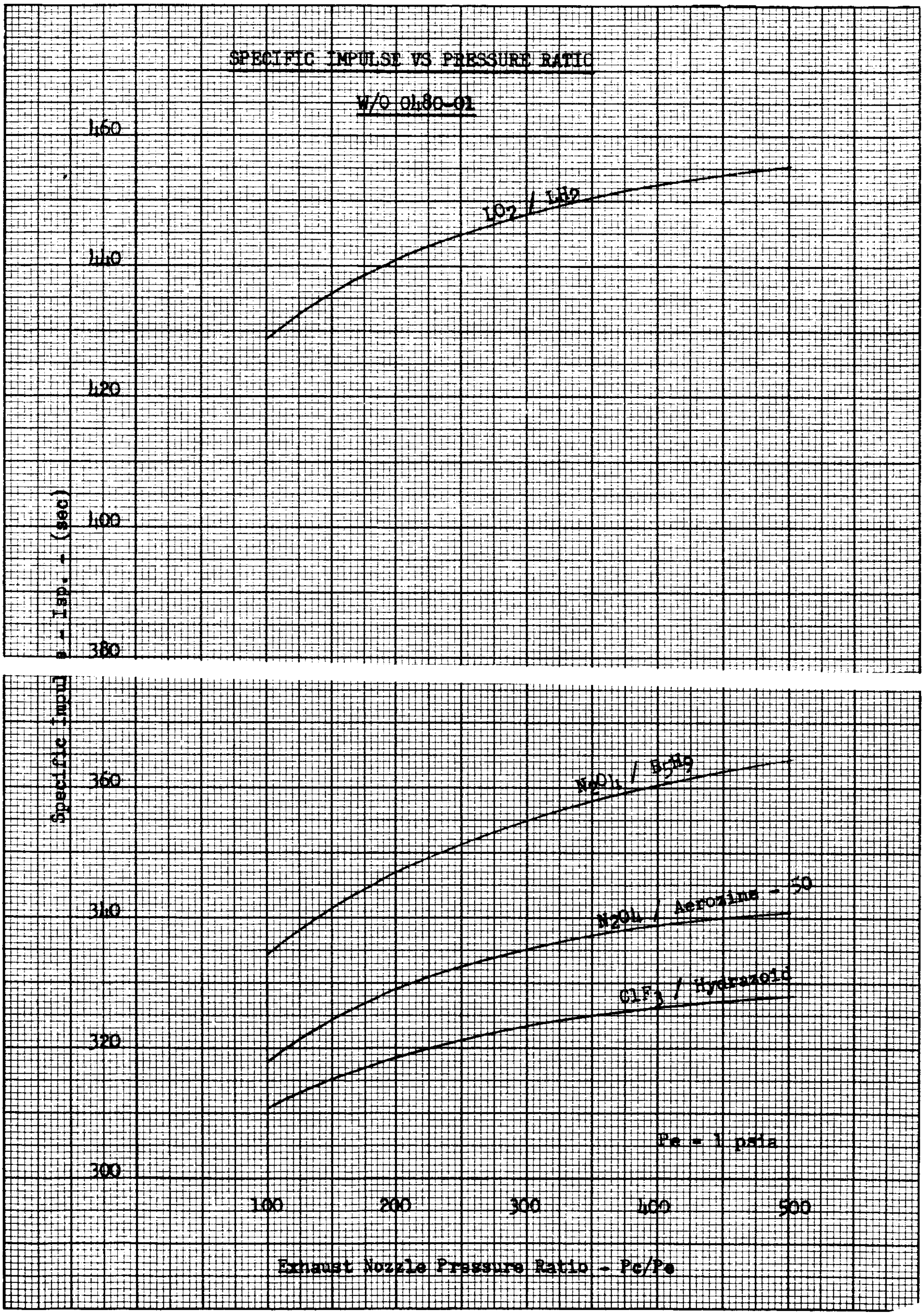

EXHAUST NOZZLE
AREA RATIO VS PRESSURE RATIO
W/O 0480-01
60
40
30
20
10
100
200
300
400
500
N2O4 / B5 H9
N2O4 / Aerozine - 50
LO2 / LH2
ClF3 / Hydrazoid
Pa = 1 psia
Exhaust Nozzle Pressure Ratio - Pc/Pa

Figure IV-2


SPECIFIC IMPULSE VS PRESSURE RATIO
W/O 0480-01
460
440
420
400
380
360
340
320
300
Specific Impulse - Isp. - (sec)
LO_2 / LH_2
N_2O_4 / B_5H_9
N_2O_4 / Aerozine - 50
ClF_3 / Hydrazoid
Pe = 1 psia
100
200
300
400
500
Exhaust Nozzle Pressure Ratio - Pc/Pe

SUBJECT Flow Diagram for System Weight Determination

System Weight Determination

Stored Gas Systems

Cold Gas
Heated Gas
Line Heated
Bladder Systems

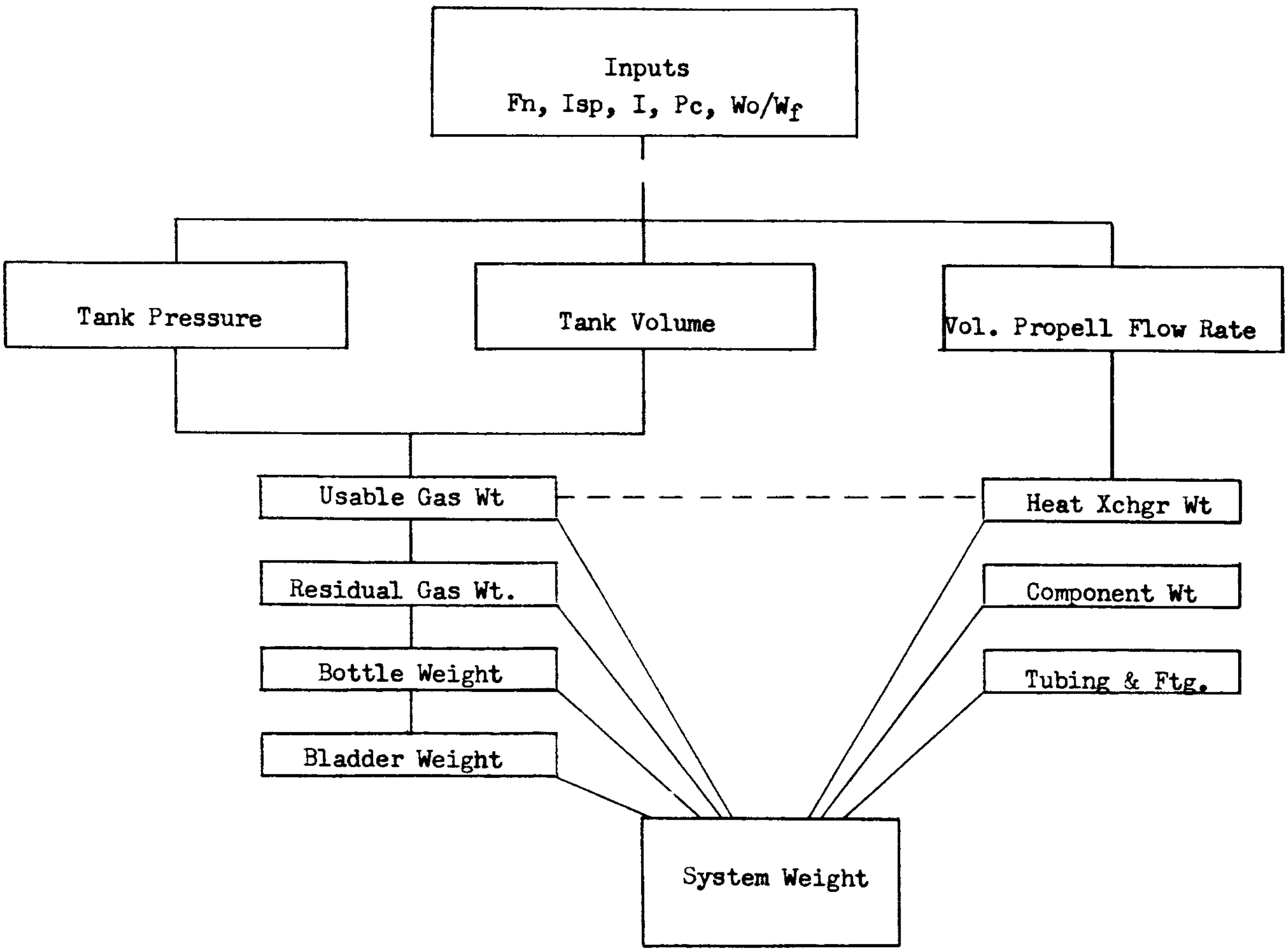
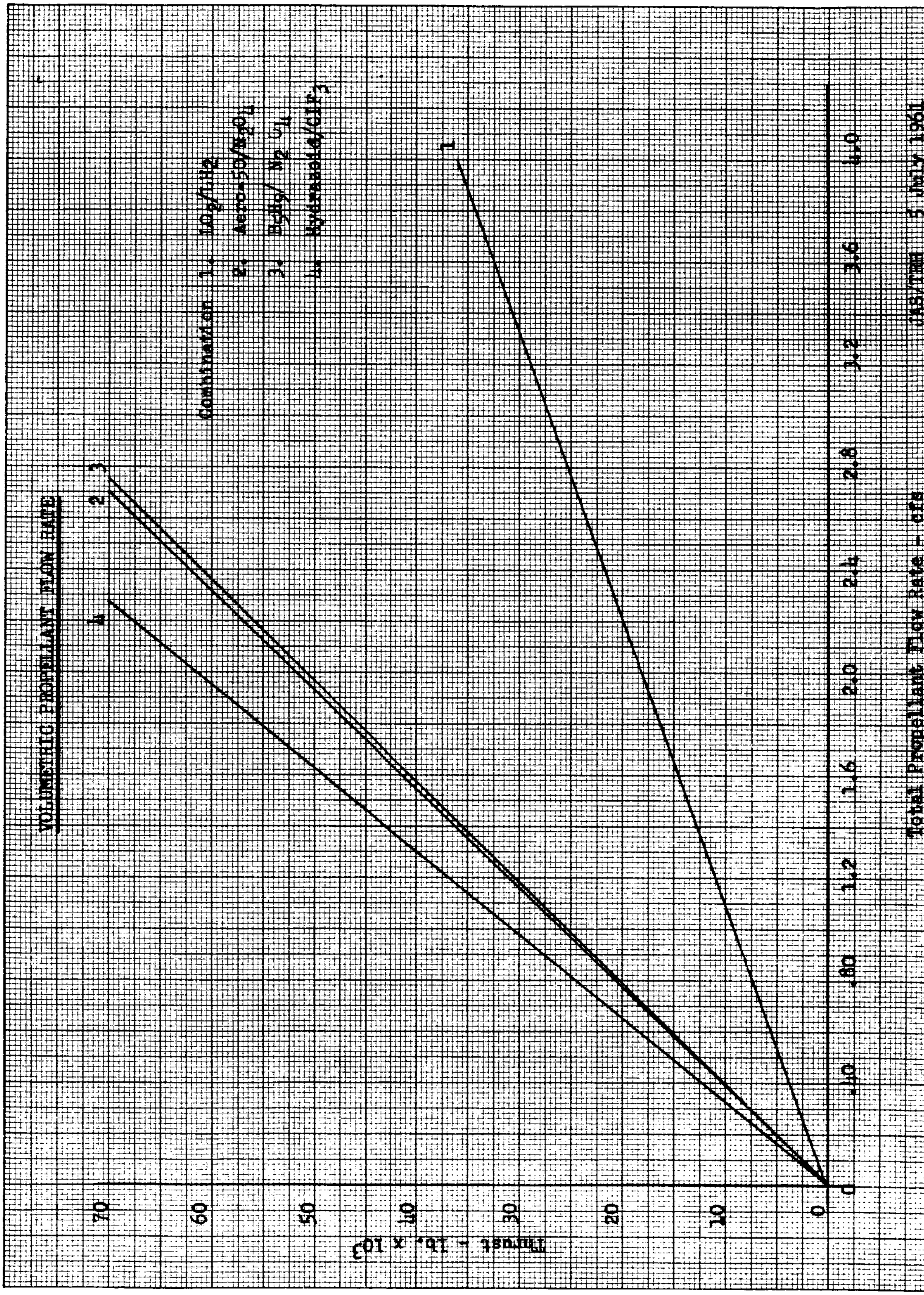


Figure IV-4


VOLUMETRIC PROPELLANT FLOW RATE
Combination 1. lO_2/lH_2
2. Aero-50/N_2O_4
3. B_5H_9/ N_2H_4
4. Hydrazoid/ClF_3
1
2
3
4
Thrust - lb. x 10^3
70
60
50
40
30
20
10
0
0
.40
.80
1.2
1.6
2.0
2.4
2.8
3.2
3.6
4.0
Total Propellant Flow Rate - cfs
JAS/TBH 5 July 1961

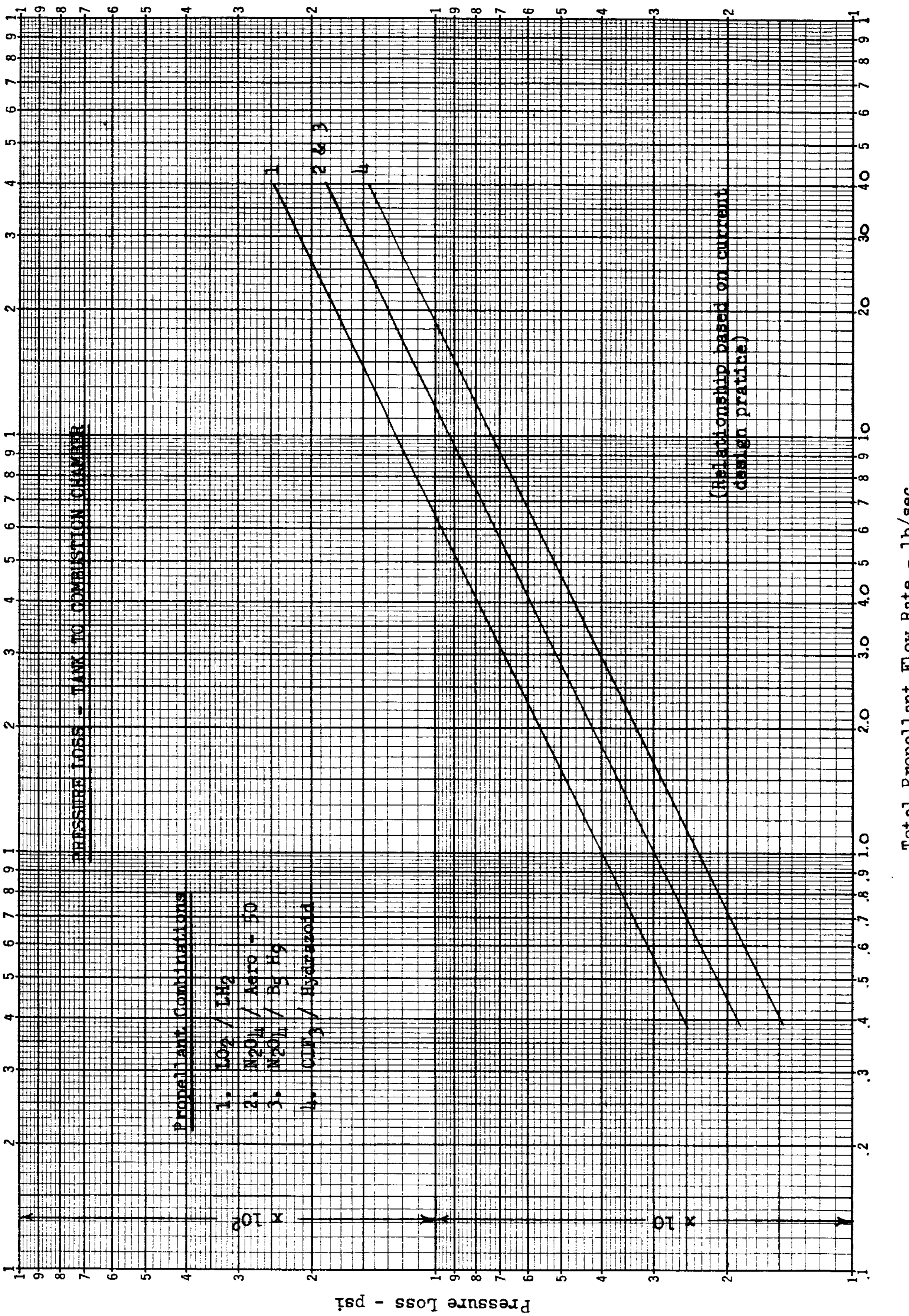

PRESSURE LOSS - TANK TO COMBUSTION CHAMBER
Propellant Combinations
1. LO_2 / LH_2
2. N_2O_4 / Aero - 50
3. N_2O_4 / B_5H_9
4. ClF_3 / Hydrazoid
(Relationship based on current design pratice)
1
2 & 3
4
x 10
x 10
Total Propellant Flow Rate - lb/sec
Pressure Loss - psi

Figure IV-6

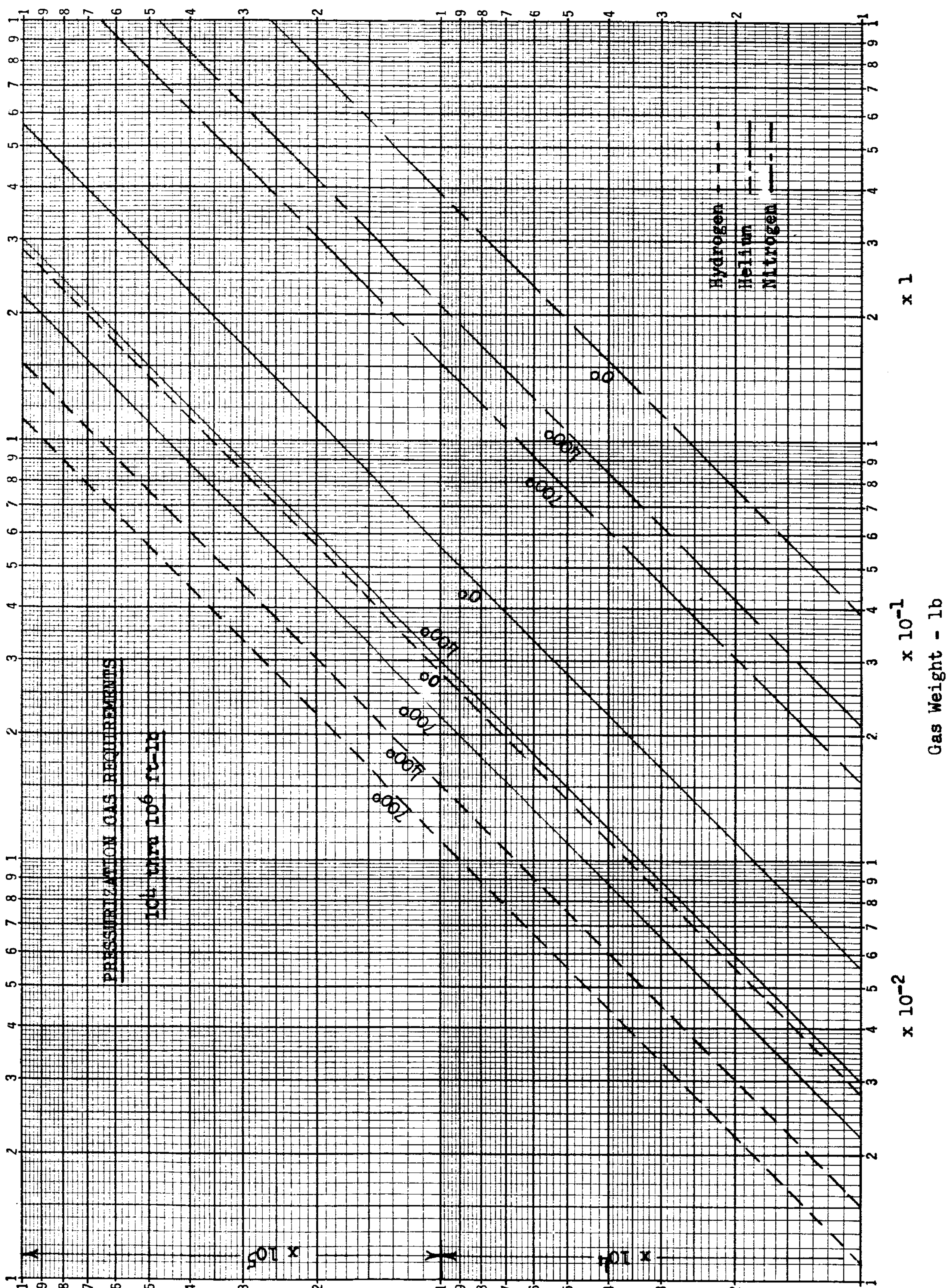

PRESSURIZATION GAS REQUIREMENTS
10⁴ thru 10⁶ ft-lb
Hydrogen
Helium
Nitrogen
x 1
x 10⁻¹
x 10⁻²
Gas Weight - lb
x 10⁵
x 10⁴
Pressurization Work (PV) - ft - lb

Figure IV-7

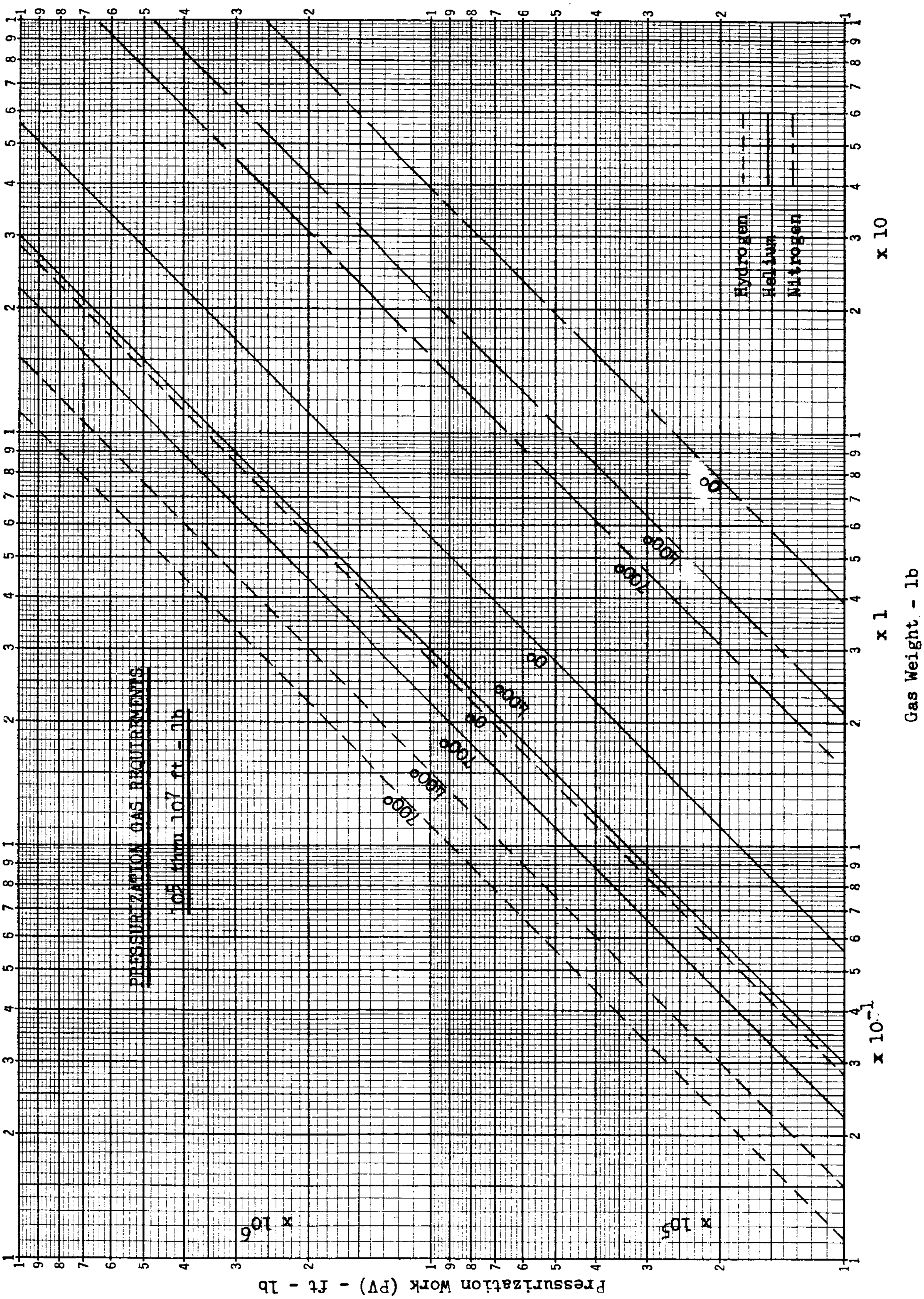

PRESSURIZATION GAS REQUIREMENTS
10^5 thru 10^7 ft - lb
Hydrogen
Helium
Nitrogen
Gas Weight - lb
x 10
x 1
x 10^{-1}
Pressurization Work (PV) - ft - lb
x 10^6
x 10^5
1000
0

Figure IV-8

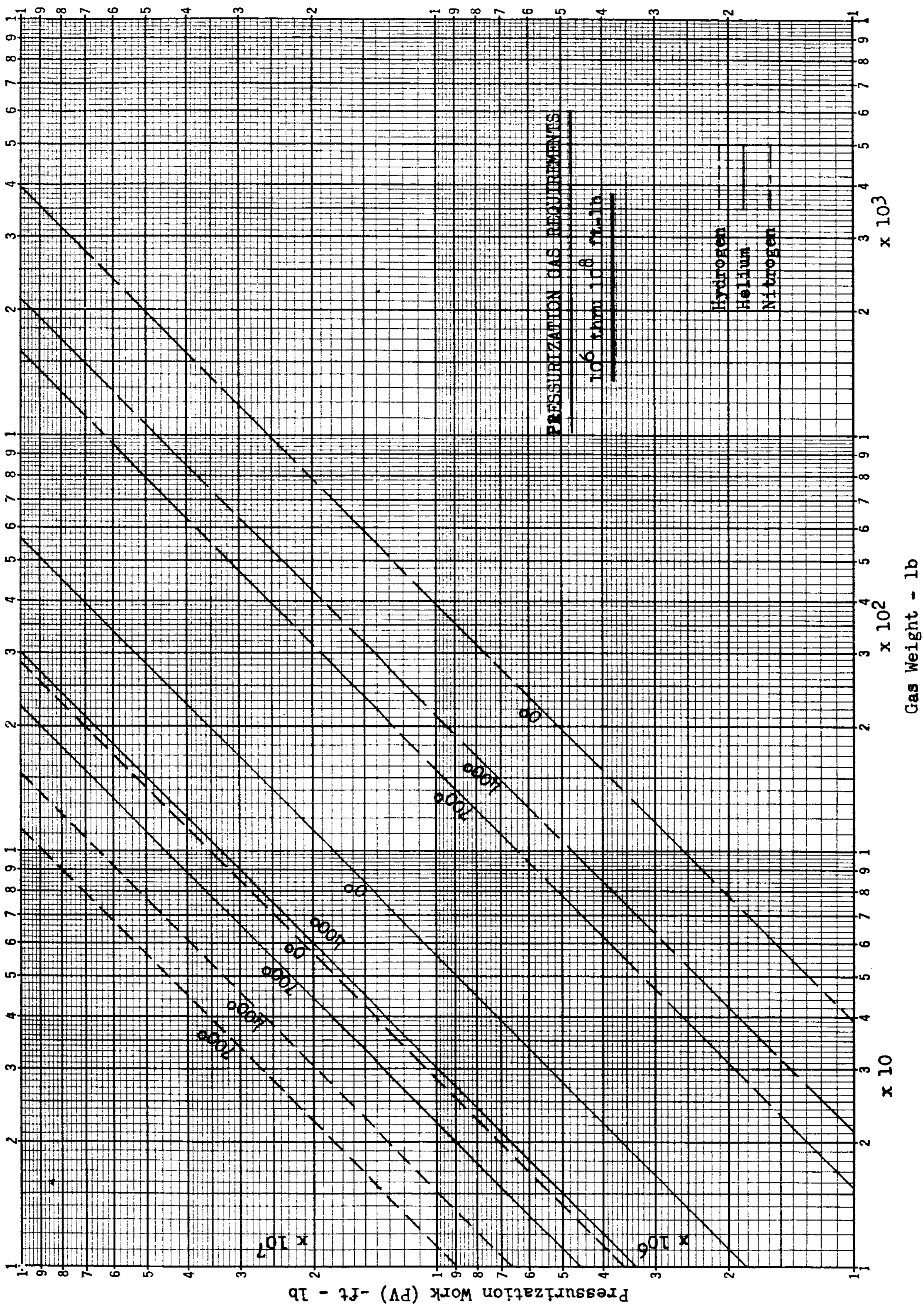

PRESSURIZATION GAS REQUIREMENTS
10^6 thru 10^8 ft-lb
Hydrogen
Helium
Nitrogen
Gas Weight - lb
x 10^3
x 10^2
x 10
Pressurization Work (PV) -ft - lb
x 10^7
x 10^6
0°
1000°

Figure IV-9

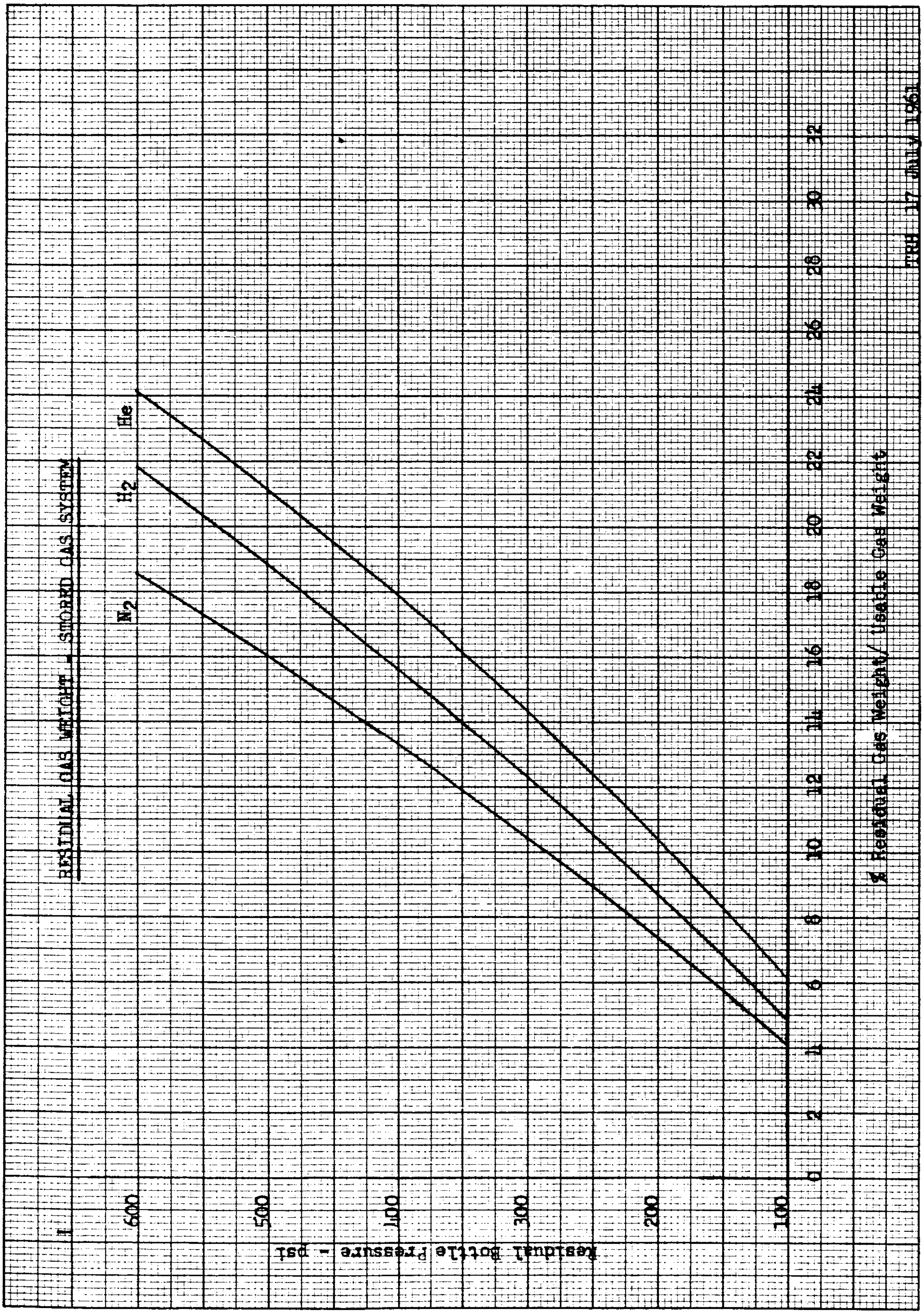

RESIDUAL GAS WEIGHT - STORED GAS SYSTEM
He
H_2
N_2
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
30
32
% Residual Gas Weight/ Usable Gas Weight
600
500
400
300
200
100
Residual Bottle Pressure - psi
TRH 17 July 1961

Figure IV-10

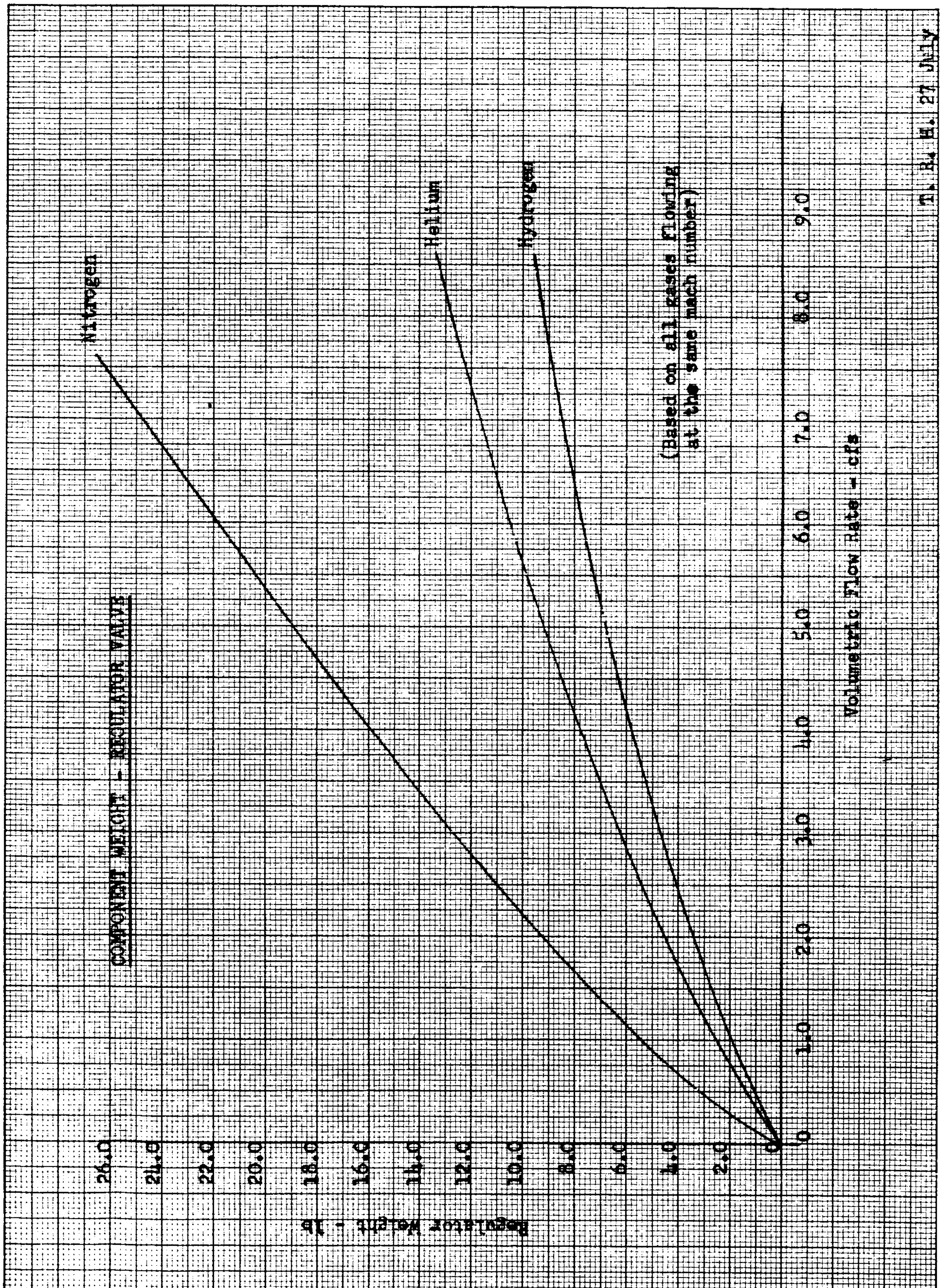

COMPONENT WEIGHT - REGULATOR VALVE
Nitrogen
Helium
Hydrogen
(Based on all gases flowing at the same mach number)
Volumetric Flow Rate - cfs
Regulator Weight - lb
0 1.0 2.0 3.0 4.0 5.0 6.0 7.0 8.0 9.0
0 2.0 4.0 6.0 8.0 10.0 12.0 14.0 16.0 18.0 20.0 22.0 24.0 26.0
T. B. H. 27 July

Figure IV-11

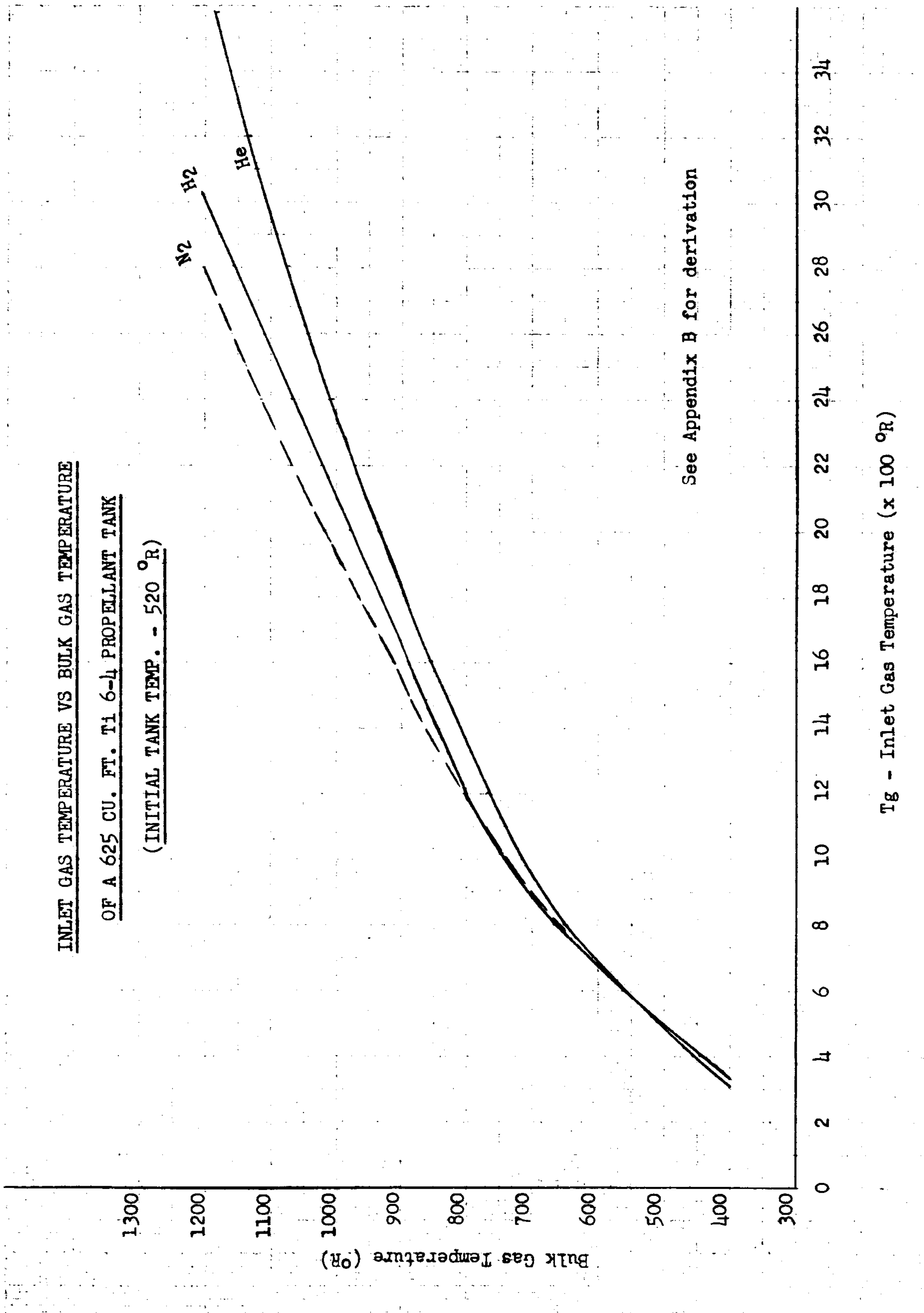

INLET GAS TEMPERATURE VS BULK GAS TEMPERATURE
OF A 625 CU. FT. Ti 6-4 PROPELLANT TANK
(INITIAL TANK TEMP. - 520 °R)
N2
H2
He
See Appendix B for derivation
Tg - Inlet Gas Temperature (x 100 °R)
Bulk Gas Temperature (°R)
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
30
32
34
300
400
500
600
700
800
900
1000
1100
1200
1300

Figure IV-12

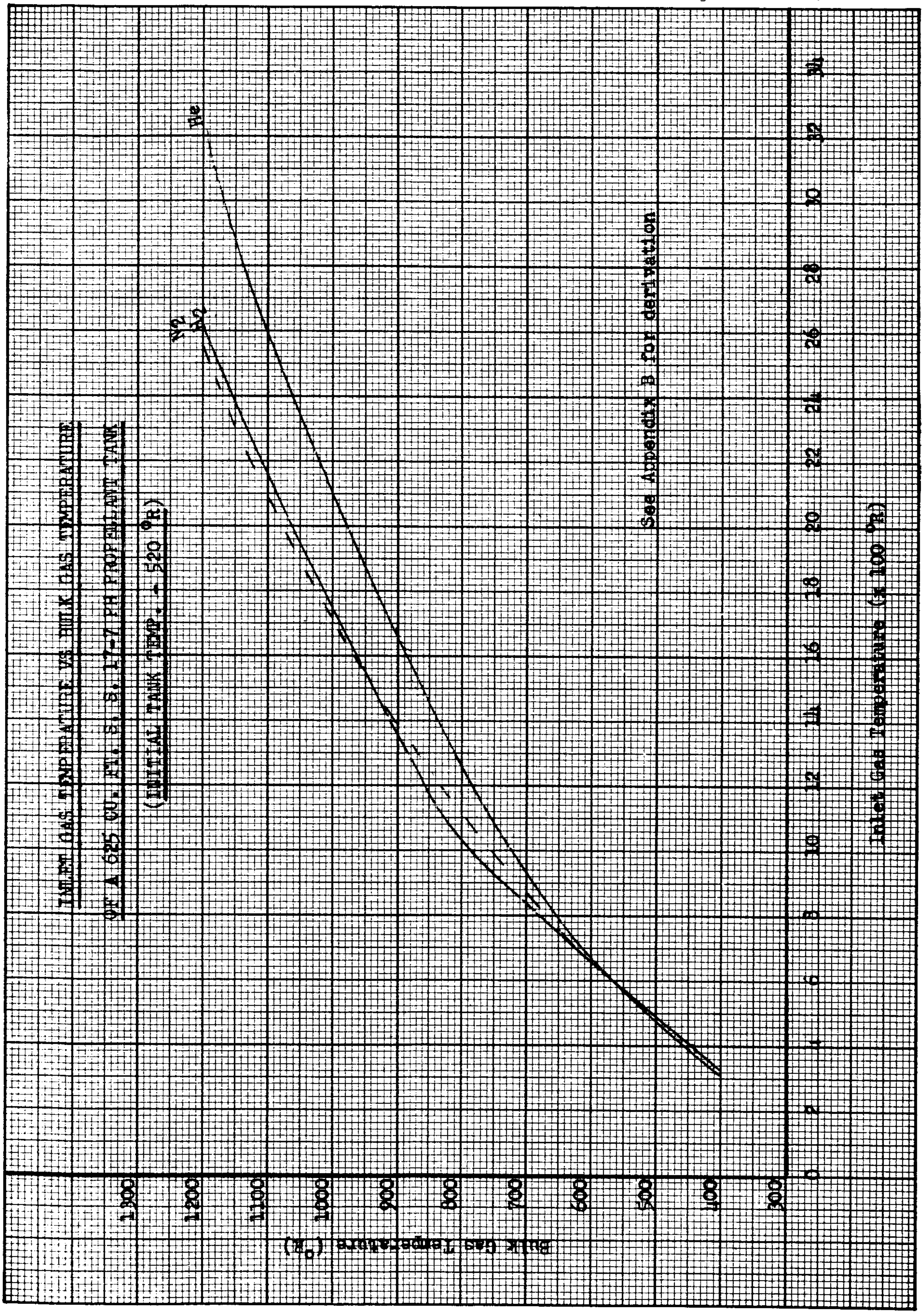

INLET GAS TEMPERATURE VS BULK GAS TEMPERATURE
OF A 625 CU. FT. S. S. 17-7 PH PROPELLANT TANK
(INITIAL TANK TEMP. = 520 °R)
See Appendix B for derivation
He
N2
H2
Inlet Gas Temperature (x 100 °R)
Bulk Gas Temperature (°R)

Figure IV-13

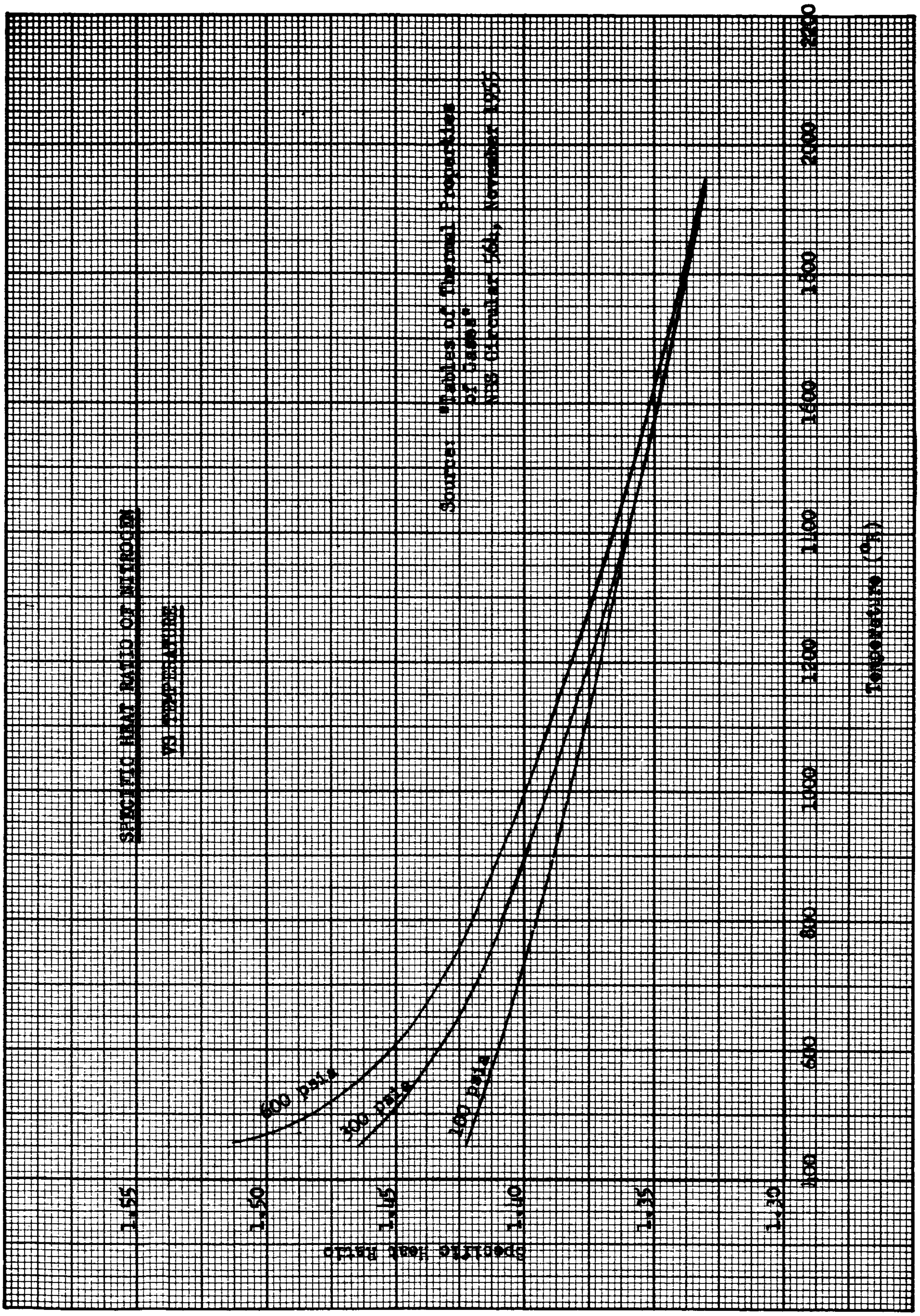

SPECIFIC HEAT RATIO OF NITROGEN
VS TEMPERATURE
Source: "Tables of Thermal Properties of Gases"
NBS Circular 564, November 1955
600 psia
300 psia
100 psia
Specific Heat Ratio
1.55
1.50
1.45
1.40
1.35
1.30
Temperature (°F)
400
600
800
1000
1200
1400
1600
1800
2000
2200

Figure IV-14

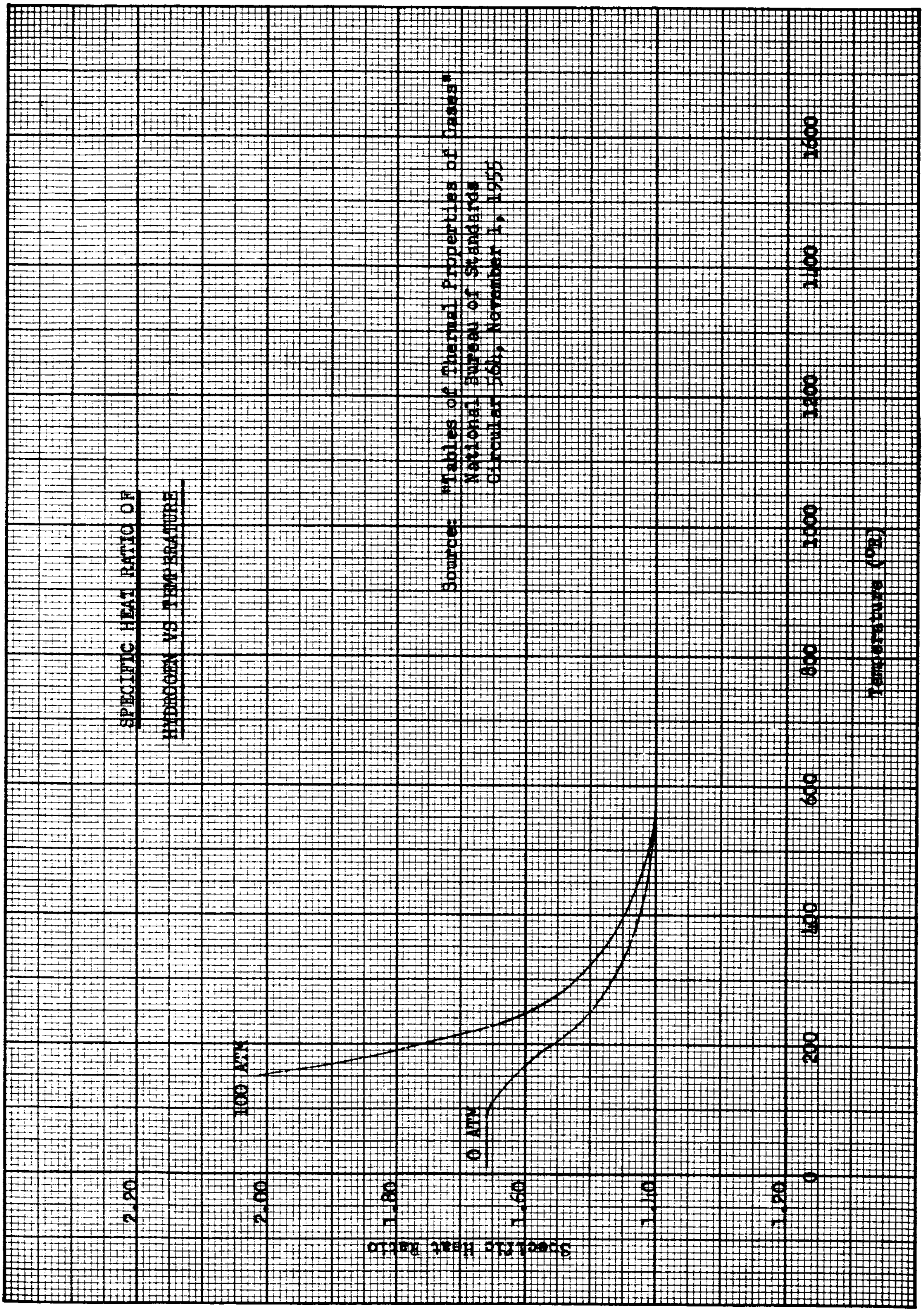

SPECIFIC HEAT RATIO OF
HYDROGEN VS TEMPERATURE
Source: "Tables of Thermal Properties of Gases"
National Bureau of Standards
Circular 564, November 1, 1955
100 ATM
0 ATM
2.20
2.00
1.80
1.60
1.40
1.20
Specific Heat Ratio
0
200
400
600
800
1000
1200
1400
1600
Temperature (°R)

Figure IV-15

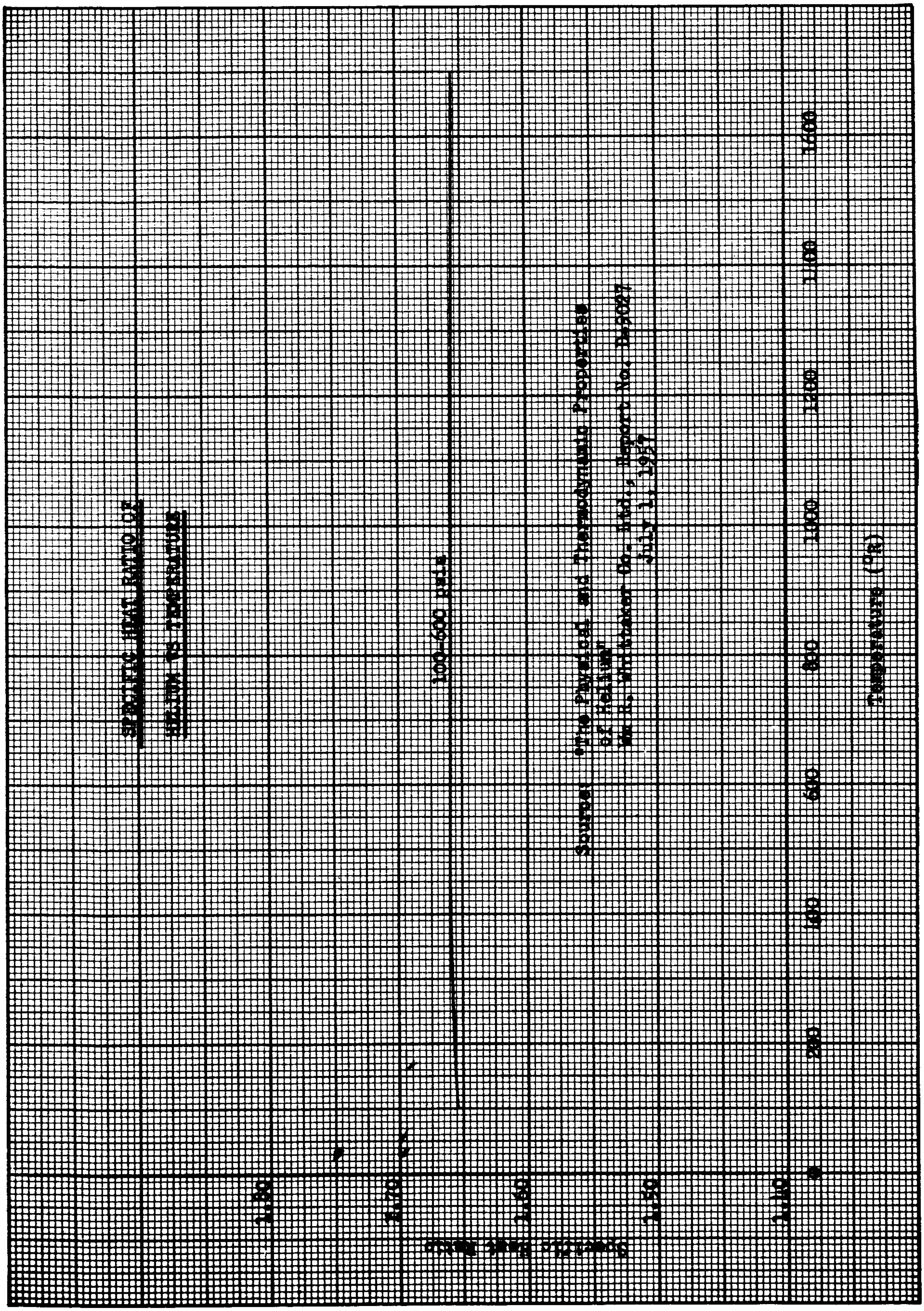

SPECIFIC HEAT RATIO OF
HELIUM VS TEMPERATURE
1.80
1.70
1.60
1.50
1.40
Specific Heat Ratio
100-600 psia
Source: "The Physical and Thermodynamic Properties
of Helium"
Wm R. Whitaker Co. Ltd., Report No. D-9027
July 1, 1957
0
200
400
600
800
1000
1200
1400
1600
Temperature (°R)

Figure IV-16

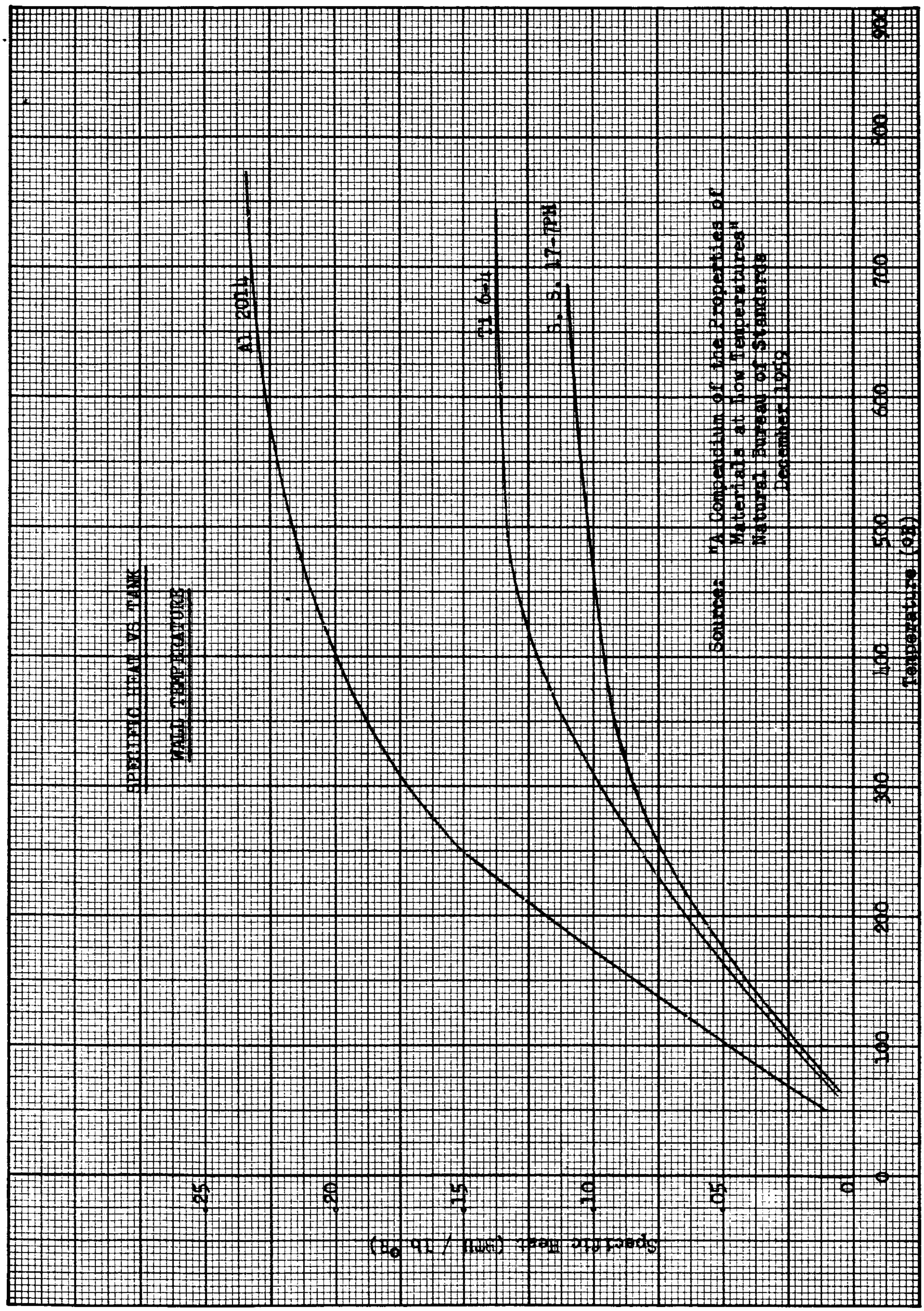

SPECIFIC HEAT VS TANK
WALL TEMPERATURE
Al 2014
Ti 6-4
S. S. 17-7PH
Source: "A Compendium of the Properties of
Materials at Low Temperatures"
Natural Bureau of Standards
December
Temperature (°R)
Specific Heat (BTU / lb °R)
0
100
200
300
400
500
600
700
800
900
.05
.10
.15
.20
.25

Figure IV-17

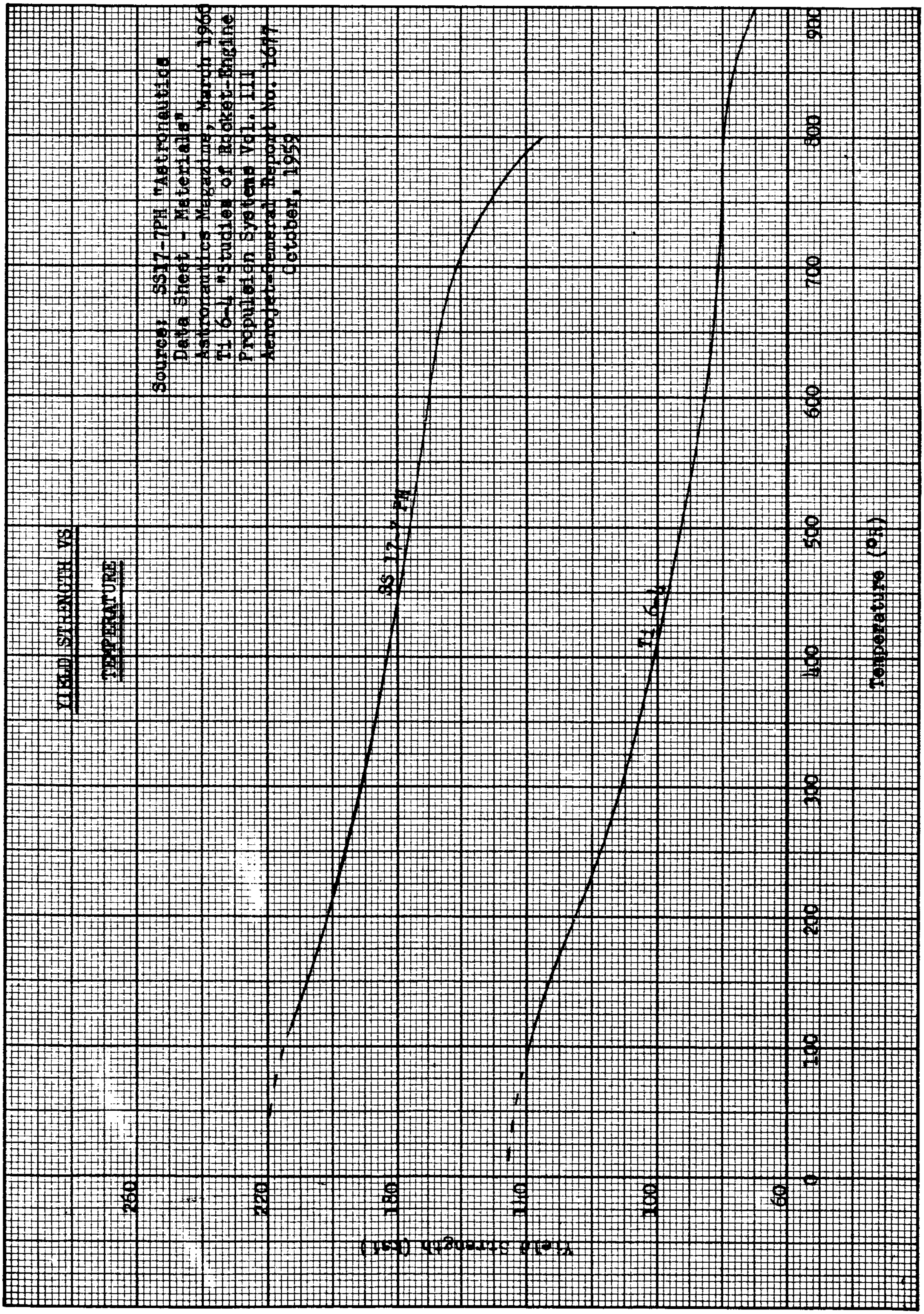

YIELD STRENGTH VS
TEMPERATURE
Source: SS17-7PH "Astronautics Data Sheet - Materials" Astronautics Magazine, March 1960
Ti 6-4 "Studies of Rocket-Engine Propulsion Systems Vol. III Aerojet-General Report No. 1677 October, 1959
SS 17-7 PH
Ti 6-4
Temperature (°F)
0
100
200
300
400
500
600
700
800
900
Yield Strength (ksi)
260
220
180
140
100
60

Figure IV-18

V. MAIN TANK INJECTION TESTS

A. GENERAL

Main tank injection (MTI) is a method of pressurization wherein the pressurizing gases are generated by a controlled reaction taking place in the tank to be pressurized. If a propellant combination is hypergolic, the fuel tank may be pressurized by injection of oxidizer and the oxidizer tank by injection of fuel. Non-hypergolic propellant combinations may be pressurized with MTI by introducing other substances which are hypergolic with the propellants being pressurized. The reaction occurring in MTI and, consequently, the properties of the pressurizing gases may be varied by changing the method of injection. The two basic types of injection are top-surface and subsurface. In top-surface injection the fuel or oxidizer is injected into the tank ullage and the reaction takes place at the liquid surface. Heat transfer to the liquid is at a minimum and ullage gas temperatures are quite high. In subsurface injection the fuel or oxidizer is injected below the liquid surface and the reaction takes place as the gaseous combustion products rise to the liquid surface. Heat transfer to the liquid is higher than in top-surface injection and ullage gas temperatures are much lower.

An analysis of top-surface injection based on experimental results has been performed, and the results are included in the design guide.

No experimental results were available for subsurface main tank injection with the fuel combinations considered in this report. Therefore, a test program was conducted to determine the feasibility of the use of subsurface MTI pressurization with Aerozine-50 (50% by weight hydrazine and UDMH) and N_2O_4. This propellant combination was chosen for experiment because it is hypergolic and storable, and because preliminary analysis indicated the desireability of MTI pressurization if the molecular weight of the combustion products was sufficiently low.

Twelve tests were made to demonstrate the feasibility of pressure-expelling Aerozine-50 by the direct subsurface injection of N_2O_4 into the Aerozine propellant tank. Of these tests, six produced useful data. Several other tests did not produce useful data due to injector plugging or valve malfunctions.

[redacted]

B. TEST APPARATUS

The test apparatus for the subsurface injection of N_2O_4 into a tank of Aerozine-50 consisted of an N_2O_4 tank at constant pressure, an injector with an orifice to limit N_2O_4 flow, a 1.3 gal tank for Aerozine-50, gas sample bombs, and associated valves and instrumentation.

Figure V-1 is a schematic of the system. The N_2O_4 tank pressure was maintained at a pre-determined value by means of a dome-loaded pressure regulator. The N_2O_4 was filtered through a fine-mesh, stainless steel screen and its flow rate was measured by the pressure differential across a sharp-edged orifice.

The injector was fabricated by welding a 12-in. length of 0.0225 in. ID capillary tubing to the inside of a 1/8-in. stainless tube which provided rigidity for the assembly. The injector end of the capillary tube was welded shut and a 0.0135-in. hole was drilled through the weld into the capillary tubing. A 0.005-in.-dia wire was inserted and the hole was peened until it was approximately 0.005 to 0.006 in. in diameter, Figure V-2. The other end of the assembly was connected to a check valve to prevent Aerozine-50 from entering the injector. The injector was held at the bottom of the Aerozine-50 tank by a "swedge" lock fitting which allowed the tip of the injector to be raised or lowered as desired.

The volume of the Aerozine-50 tank was 1.31 gal (0.175 ft^3) with the volume of tubing and other fittings directly connected to the tank providing an additional 0.07 gal (0.009 ft^3). The approximate inside height of the welded steel tank was 15-1/2 in. Figure V-3 shows the Aerozine tank installed in the test system.

Aerozine-50 flow was measured by a Potter meter and discharged through a restricting orifice to the atmosphere. Three thermocouples were used to measure the temperatures in the Aerozine tank. They were placed 3.5, 7, and 14 in. from the bottom of the tank and extended 2.5 in. into the tank. A fourth thermocouple measured the temperature of the Aerozine as it was expelled from the tank. The tank pressure and other instrumentation were recorded on an oscillograph.

Two gas samples were simultaneously obtained for analysis by the use of solenoid-operated valves. Due to the relatively large volume of the gas samples compared with the Aerozine-50 tank, these samples were taken immediately following the expulsion, with the exception of Run 9D.

As a safety precaution, a burst diaphragm set to rupture at 810 psi was located at the top of the Aerozine tank. In case of a buildup from the operating pressure of 400 psi, a high-pressure microswitch was set to close the N_2O_4 valve and to actuate a solenoid-operated vent valve. This valve was also used to relieve the pressure in the Aerozine tank after each test.

It should be noted that the sum of the pressure drops from the N_2O_4 tank to the Aerozine tank and from the Aerozine tank to atmospheric pressure is constant throughout a run. Since N_2O_4 flow is dependent on the latter, an increase in Aerozine tank pressure and flow rate requires a drop in N_2O_4 flow rate and vice versa.

C. PROCEDURE

The N_2O_4 and Aerozine tanks were pressurized with helium to approximately 490 psia and 380 psia, respectively, before each run. In the earlier tests, both the N_2O_4 and Aerozine flow were initiated by solenoid-valve actuation, but it proved difficult to obtain the proper valve-opening synchronization. This problem was eliminated by adding a burst diaphragm, set for 410 psi, in the Aerozine discharge line. As before, both valves were opened but the burst diaphragm prevented Aerozine flow until the N_2O_4 brought the tank pressure up to 410 psi. With both propellants flowing, the system approached equilibrium operation within a few seconds.

When the Aerozine Potter meter indicated the completion of liquid expulsion, the Aerozine valve was closed and gas samples were taken of the remaining pressurizing gas. The system was then vented to complete the run. Several liquid samples of expelled Aerozine were taken and analyzed. In some cases, the Aerozine was expelled as much as three times, with samples taken after each expulsion.

D. EXPERIMENTAL RESULTS

The six tests which yielded useful data were Run Nos. 3,6,7,9B, 9C, and 9D.

1. Run No. 3

During this test there was no burst diaphragm in the Aerozine discharge line. The Aerozine tank was filled with 1.18 gal of the fuel. The injector was located 3.5 in. from the bottom of the tank and the tank was pressurized with helium to 420 psia. Pressure on the N_2O_4 tank was maintained at 495 psia. A malfunction delayed the opening of the N_2O_4 valve, and the pressure on the Aerozine tank dropped to 200 psia before contact with the N_2O_4 began. The pressure had risen to 365 psia by the time the unit was shut down. The N_2O_4 flow was high and the data indicated a surging of the flow. The volume of Aerozine expelled was 0.674 gal over a period of 14.3 sec.

2. Run No. 6

The data from this test are shown in Figure V-4. The injector orifice was reworked to give a flow of about 0.003 lb/sec with 100 psi pressure differential between the two tanks. The volume of Aerozine-50 placed in the Aerozine tank was 1.19 gal. The Aerozine tank was pressurized to 395 psia with helium, and N_2O_4 tank pressure was maintained at 505 psia. After breaking the Aerozine flow burst diaphragm, the tank pressure decreased to 390 psia and then rose to 470 psia before dropping to 390 psia at the end of the expulsion. The Aerozine flow rate varied $\pm$ 7.3% from a nominal of 0.371 lb/sec.

3. Run No. 7

The data from this test are shown in Figure V-5. The injector probe was lowered to 1/2 in. from the bottom of the Aerozine tank. A volume of 1.19 gal. of Aerozine-50 (reused from Run No. 6) was placed in the Aerozine tank. The Aerozine tank was pressurized with helium to 395 psia and 505 psia was maintained on the N_2O_4 tank. The pressure on the Aerozine tank rose to 430 psia then dropped to 320 psia after the blowout disk ruptured. The pressure gradually increased to 425 psia when the unit was shut down. The volume of expelled Aerozine was 0.919 gal during 23.8 sec. The N_2O_4 injector was partly plugged during

[redacted]

the run causing somewhat erratic N_2O_4 flow, but the Aerozine flow was not sensitive to it and varied only 0.358 lb/sec ± 3.9%.

The same trends in flow rates, temperatures and tank pressure can be seen as were apparent in Run 6, but the lower position of the injector tip did not allow as much of the trend to be completed.

4. Run 9B

The data from this test are shown in Figure V-6. The injector probe was placed 1/2 in. from the bottom of the Aerozine tank and 1.23 gal of Aerozine-50 were placed in the tank. The Aerozine tank pressure was set at 375 psia, and 513 psia was maintained on the N_2O_4 tank.

The burst diaphragm broke at 445 psia. T h tank pressure dropped to a minimum of 370 psia followed by a slow rise to a maximum of 485 psia near the end of the test. The volume of expelled Aerozine was 1.14 gal during 25.3 sec. The Aerozine flow rate variation was 0.382 lb/sec ± 3.2%.

The same temperature, pressure, and flow-rate trends were apparent.

5. Run No. 9C

The data from this test are shown in Figure V-7. A volume of 1.14 gal of Aerozine (reused from Run No. 9B) was placed in the Aerozine tank. The Aerozine tank pressure was set at 345 psia, and the N_2O_4 tank pressure was maintained at 495 psia.

The data for Run 9C is quite similar to that obtained from Run 9B. The Aerozine flow was initiated by the burst diaphragm at 420 psia. Tank pressure dropped to a minimum of 370 psia and slowly climbed to a maximum of 460 psia near the end of the expulsion. As before, the N_2O_4 flow rate decreased near the end of the expulsion, but it did not drop as low as it did in the previous test (Run 9B). There was no apparent cause for the difference in final N_2O_4 flow rates between the two tests. The volume of expelled Aerozine was 1.10 gal during 25.5 sec. The Aerozine flow rate variation was 0.375 lb/sec ± 3.0%.

[redacted]

6. Run No. 9D

The data from this test are shown in Figure V-8. A volume of 1.10 gal of Aerozine (reused from Run No. 9C) were placed in the Aerozine tank. The Aerozine tank pressure was set at 360 psia, and the N_2O_4 tank was maintained at 465 psia.

Tank pressure was similar to the two previous tests but with the diaphragm breaking at 420 psia, dropping to 315 psia, and rising to 335 psia when the test was interrupted by the removal of the gas samples. The N_2O_4 flow rate peaked and dropped twice before coming up to steady-state operation. This was probably due to injector plugging as there may have been some contamination from the previous two runs. The problem appeared to be eliminated by the end of the run because the N_2O_4 flow rate responded nicely to offset the removal of gas samples. The volume of expelled Aerozine was 0.80 gal. A sample of the expelled Aerozine was obtained. The run time was 22.2 sec.

7. Liquid Analysis

Fresh and expelled Aerozine-50 analyses are given in Table V-1. The Aerozine was analyzed by the standard method according to Aerojet specifications (No. AGC-44041D). The Aerozine used in these tests was higher in N_2H_4 content than the standard composition; however, this discrepancy should not affect the data obtained. The data indicate a selective reaction of the N_2O_4 with UDMH rather than with hydrazine. The Aerozine was clear when obtained from the supply tank. After the first expulsion, the liquid became a very light amber and remained that color during successive expulsions.

8. Gas Analysis

The combustion gases were analyzed by vapor-phase chromatography, and these data are shown in Table V-II. The totals in Part A of the table are not equivalent to 100 because the Aerozine vapor did not come off the chromatographic column and the helium used to purge the sample bombs is not indicated in the results of analysis. The data from Run No. 9B were discarded because of an apparent error in the volume percent of hydrogen. In Part B, the data were adjusted to give 100%. In Part C, the volume percent of Aerozine-50 vapors that would have been present in the tank at the time of sampling have been included.

[redacted]

The amount of Aerozine vapor was calculated on the basis of the Aerozine-50 vapor pressure at the final ullage gas temperature. Using the average of the gas composition for Parts B and C of Table V-II, the molecular weight of the generated gas alone was 13.3 and the molecular weight of the overall pressurizing gas (Part C) was 14.0.

E. DISCUSSION OF TEST RESULTS

These tests indicate that a pre-pressurized Aerozine-50 fuel tank may be pressure-expelled by the subsurface injection of N_2O_4. It is likely that the Aerozine flow-rate variation may be limited to less than $\pm 3\%$ using only fixed orifices for flow control. While the injector was well submerged, the reaction held the Aerozine flow variation to ± 1 or 2%. For this small injector, approximately 4 to 6 in. of Aerozine depth was required to prevent the larger flow variation, but larger injectors will probably require somewhat larger depths.

The fuel-rich N_2O_4/Aerozine-50 reaction appears to be well suited for a main tank pressurization system. The reaction seemed to occur immediately, but it was not explosive and did not produce any detrimental contaminates or explosive residues that could be detected.

1. <u>Run No. 6</u>

The following explanation of pressure and flow variation cannot be completely substantiated by the test, but fits the existing data in a logical manner. The time periods into which the run is divided are only approximations.

The approach of the liquid surface to the injector (10 to 15 sec) affected the generated gas heat transfer. The N_2O_4 flow rate was reduced by the rising tank pressure which, in turn, was caused by (1) the higher generated gas temperature as it emerged from the liquid, and (2) the additional Aerozine vapor produced by the increasing gas temperature.

It is estimated that the liquid surface reached the injector tip between 15 and 18 sec. This is based on the change of slope of T_1, the probability that the reacting N_2O_4 stream could break through the last inch or two of liquid depth, and an estimation of the liquid level as a function of time.

[redacted]

As the tip of the injector emerged from the liquid, the N_2O_4 reacted with the accumulated fuel-rich ullage vapor for a second or two. Since the fuel-rich reaction had shifted from within the liquid to the tank ullage, more of the heat of reaction was retained by the pressurizing gas, and a minimum amount of N_2O_4 was required.

When the ullage reaction had reduced the concentration of Aerozine vapor in the upper portion of the tank, less Aerozine was in contact with the injected N_2O_4 and a more nearly oxidizer-rich reaction began to occur. Tank pressure began to drop with the more oxidizer-rich reaction because (1) more N_2O_4 per pound of gas generated was used than previously, and (2) the generated gas had a higher molecular weight (greater density) which also required additional N_2O_4 flow. These two effects were the dominant factors during that period.

For example, a change in reaction mixture ratio from 0.2 to 2.0 during the period from 18 to 24.5 seconds would more than compensate for the temperature rise and pressure decrease for that period. Temperature and pressure would change the N_2O_4 flow to 65% of its value at 18 seconds, but a change in mixture ratio from 0.2 to 2.0 would require the N_2O_4 flow rate to increase by a factor of four.

2. <u>Run Nos. 7, 9B, 9C, and 9D</u>

These runs exhibited the same temperature, pressure, and flow-rate trends which were apparent in Run No. 6. However, in these runs, the trends were not allowed to progress as far since the injector was mounted closer to the bottom. As would be expected, the discussion of Run No. 6 holds for these runs, but the Aerozine expulsion was completed before the more oxidizer-rich reaction could occur.

[redacted]

[redacted]

TABLE V-1

ANALYSIS OF LIQUID AEROZINE-50

	Wt%		
	N_2H_4	UDMH	H_2O, by Difference
Specified Composition	51.0 ± 0.8	min 47.0	1.8
New Aerozine	57.3	41.3	1.4
1st Expulsion			
Run No. 9B	57.0	41.9	1.1
2nd Expulsion			
Run No. 9C	57.8	40.3	1.9
Run No. 7*	58.0	40.1	1.9
3rd Expulsion			
Run No. 9D	58.6	39.3	2.1

*No analysis made following 1st expulsion.

[redacted]

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TABLE V-2

ANALYSIS OF GASES FROM COMBUSTION OF AEROZINE-50 AND N_2O_4

	Run No. 3	Run No. 6	Run No. 7	Run No. 9B	Run No. 9C	Run No. 9D	Average
Data as Reported - Volume %							
H_2	48.1	49.0	51.3	28.5	42.0	47.3	
N_2	25.4	28.1	29.0	24.8	27.9	26.6	
CH_4	12.7	10.6	13.8	7.7	11.0	10.6	
CO	3.3	4.8	5.9	6.0	7.7	7.8	
NH_3	0.6	0.8	4.2	3.0	3.0	0.5	
H_2O	0.32	0.34	0.45	0.33	0.14	0.0	
Total	90.42	93.64	104.65	70.33	91.74	92.8	
Values Corrected to Total 100, Volume %							
H_2	53.2	52.3	49.0		45.8	51.0	50.2
N_2	28.1	30.0	27.7		30.4	28.6	29.0
CH_4	14.0	11.3	13.2	Rejected Data	12.0	11.4	12.3
CO	3.7	5.1	5.7		8.4	8.4	6.3
NH_3	0.7	0.9	4.0		3.2	0.6	1.9
H_2O	0.3	0.4	0.4		0.2	0	0.3
Total	100.0	100.0	100.0		100.0	100.0	100.0
				Molecular weight, M_{avg} = 13.3			
Values Corrected for Aerozine-50 Vapors in Gas, Volume %							
H_2	52.8	50.6	47.7		43.7	49.5	48.8
N_2	27.9	29.1	27.0		29.0	27.8	28.1
CH_4	13.9	11.0	12.9	Rejected Data	11.5	11.1	12.1
CO	3.6	5.0	5.5		8.1	8.2	6.1
NH_3	0.7	0.8	3.9		3.1	0.5	1.8
H_2O	0.4	0.4	0.4		0.1	0	0.3
A_2-50*	0.7	3.1	2.6		4.5	2.9	2.8
Total	100.0	100.0	100.0		100.0	100.0	100.0
						M_{avg} = 14.0	

*Figured from gas temperature and A-50 vapor pressure.

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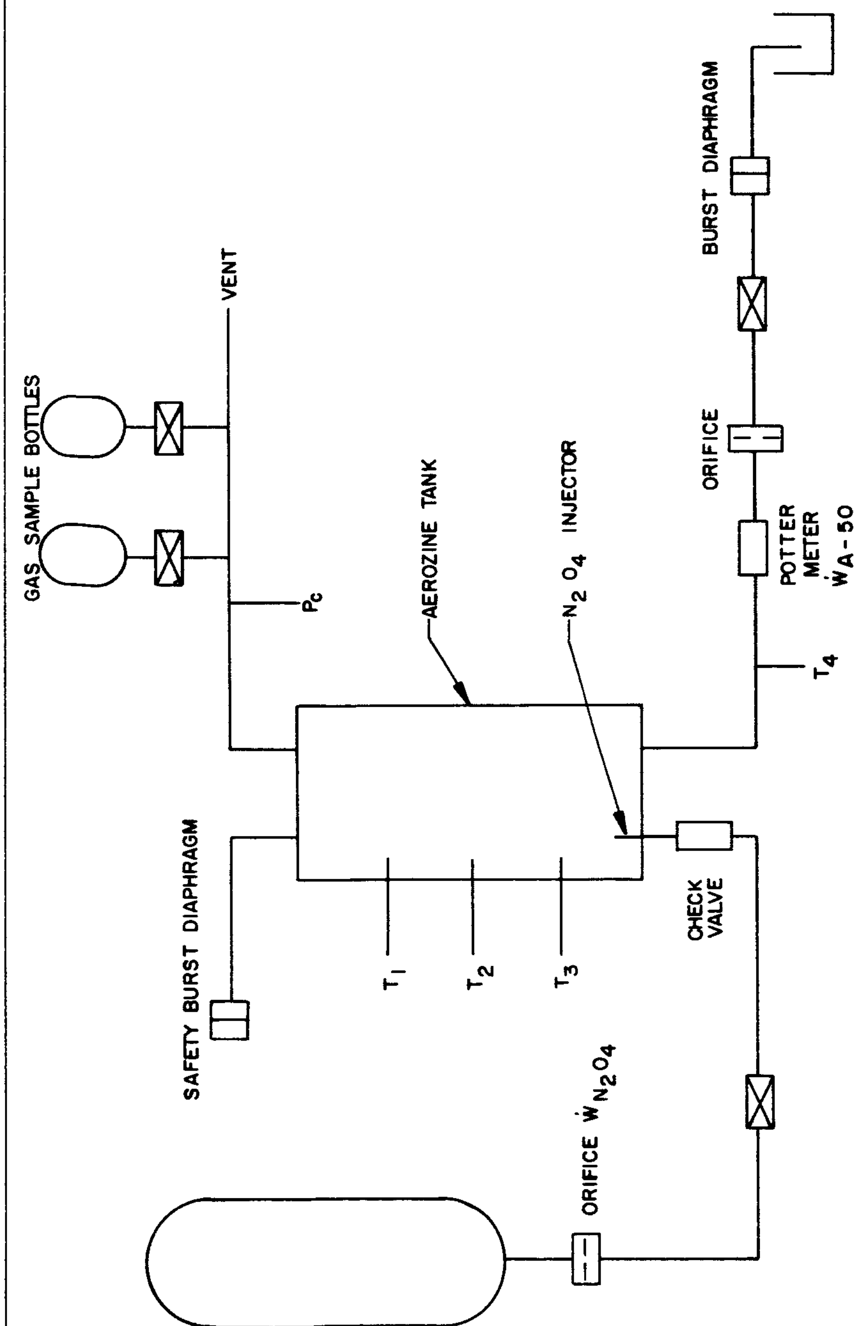
[redacted]



MAIN TANK INJECTION TEST SYSTEM SCHEMATIC

[redacted]

Figure V-1

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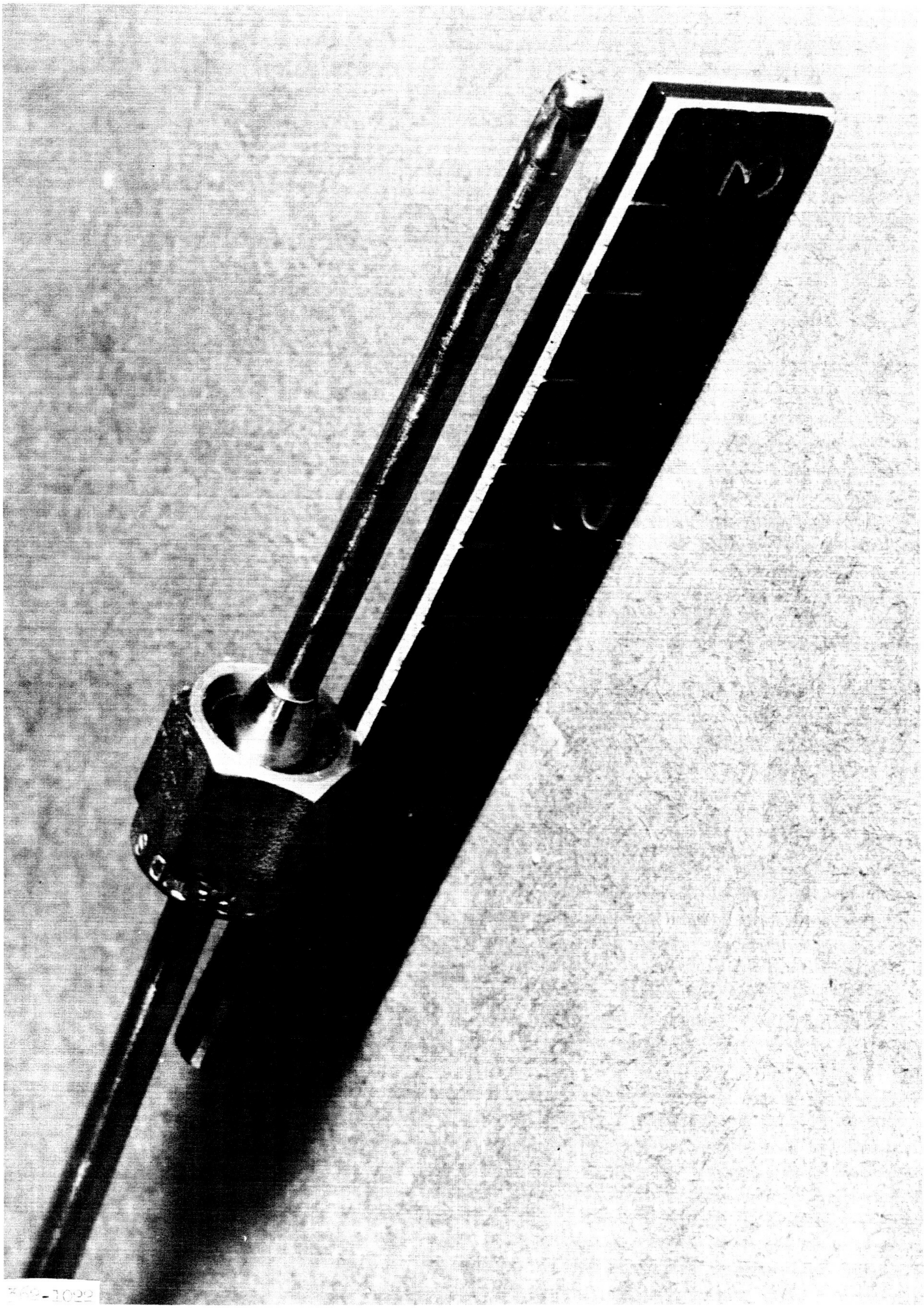

One Gallon Test Injector

[redacted]

Figure V-2

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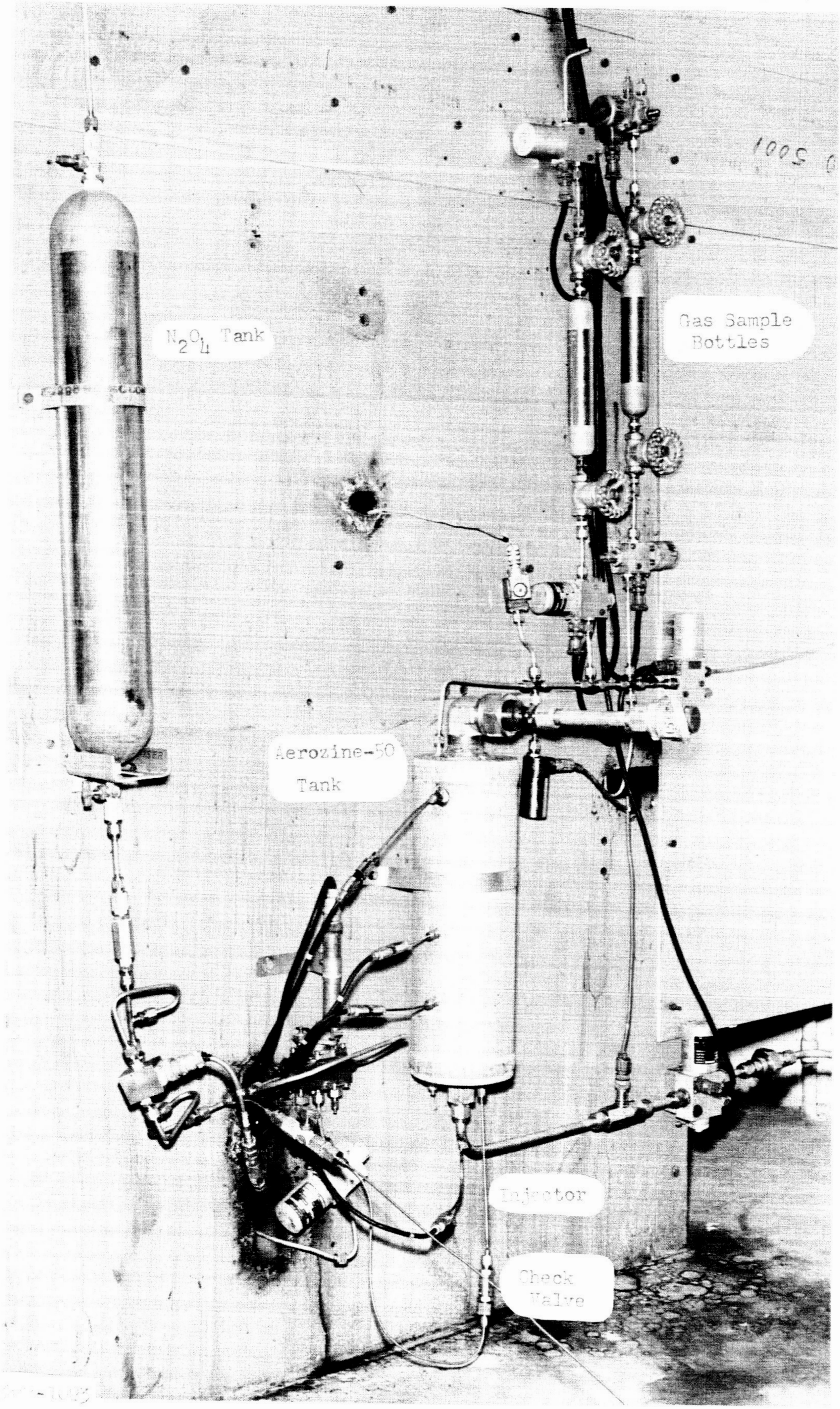

One Gallon Test Setup

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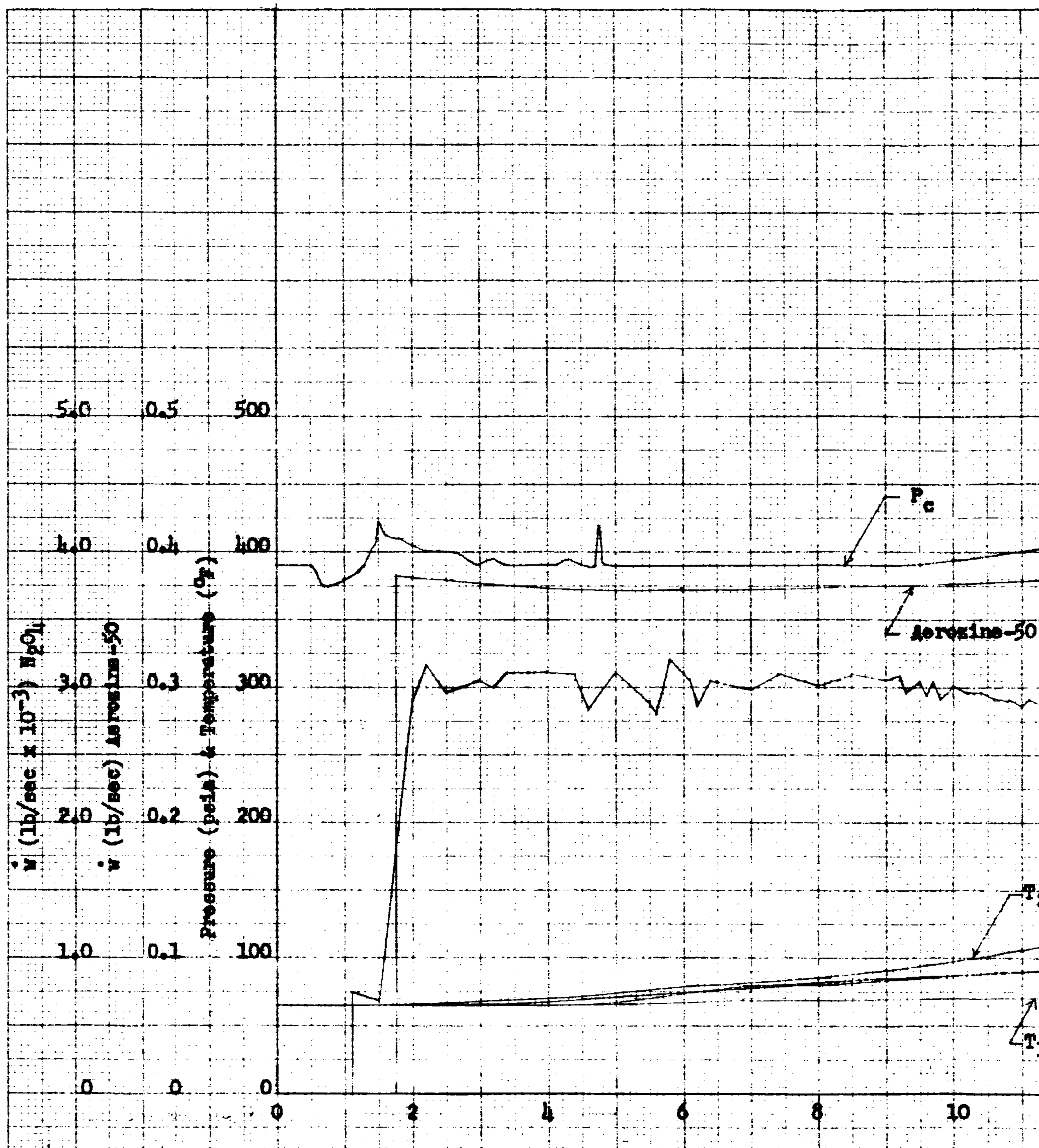
Figure V-3

[redacted]



V-4-1

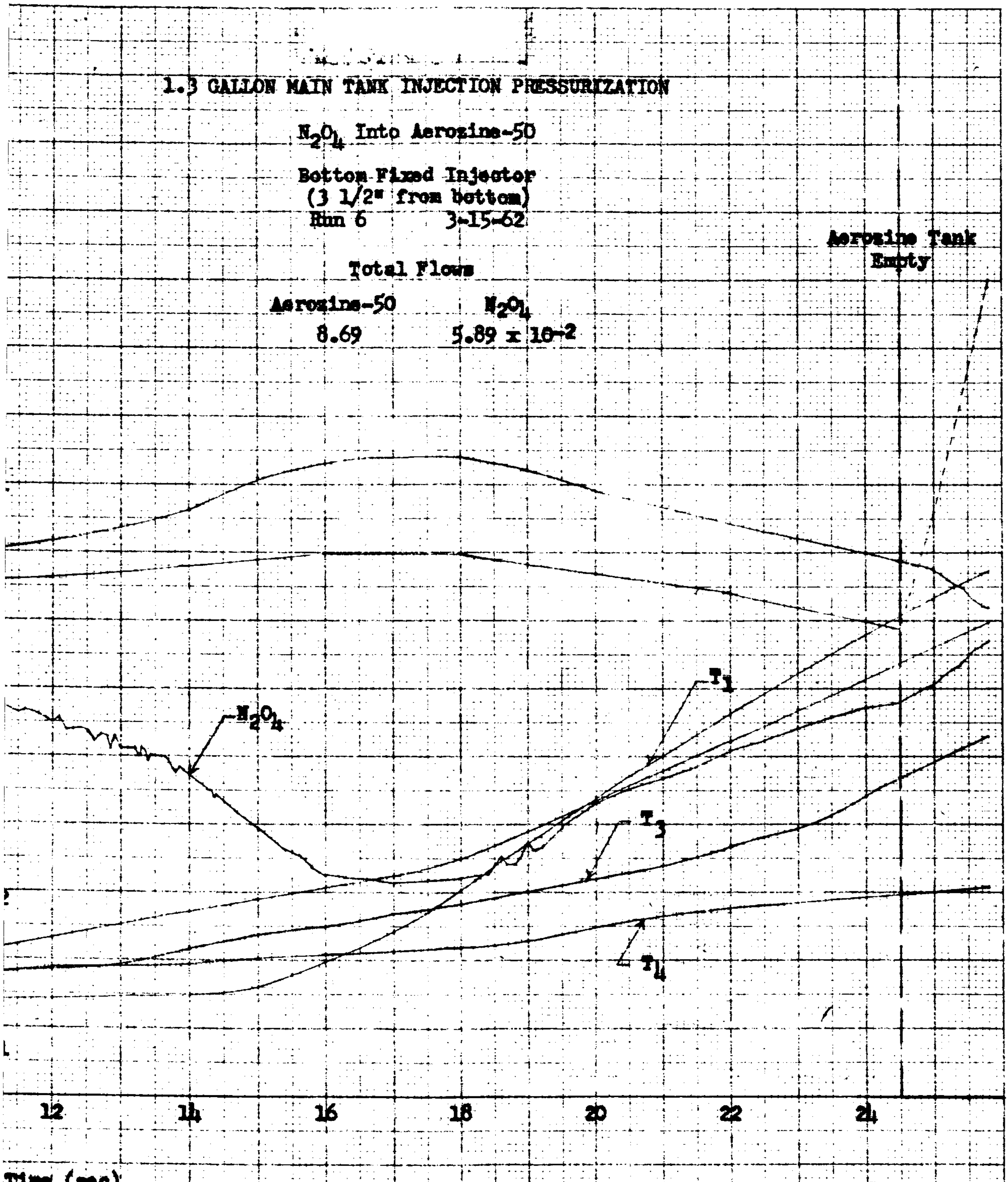


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Figure V-4 2

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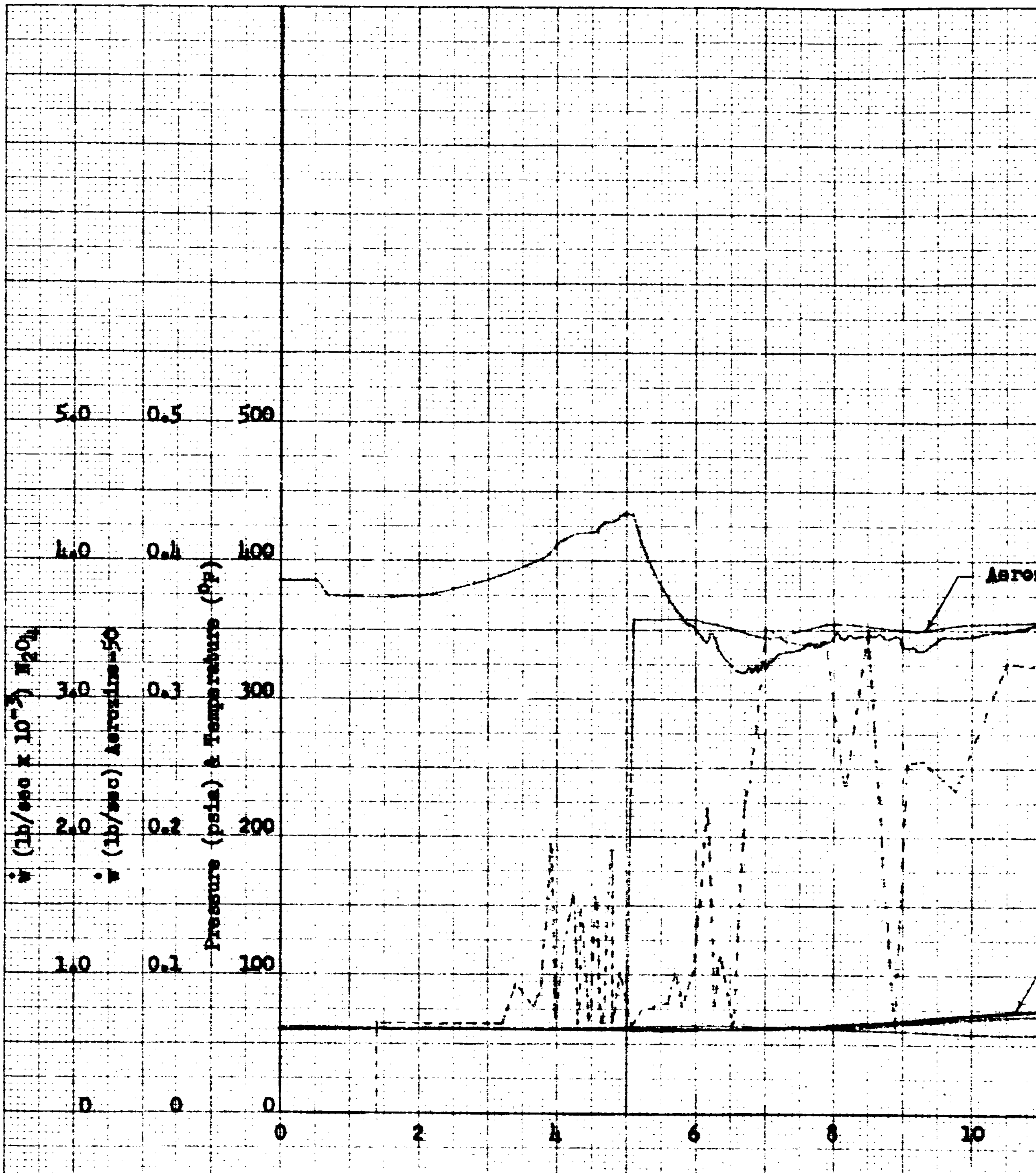

V-5-11

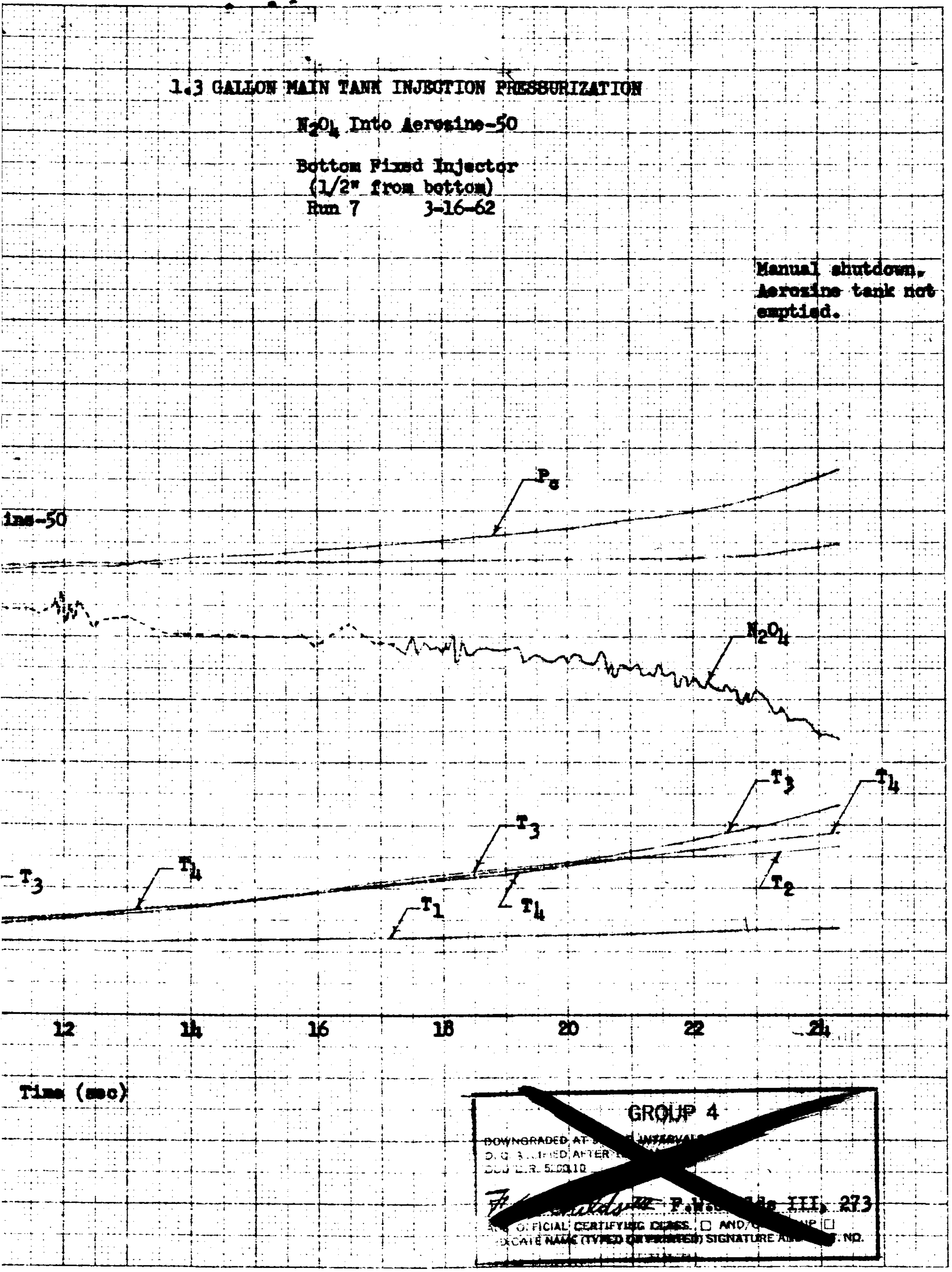

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Figure V-5

2

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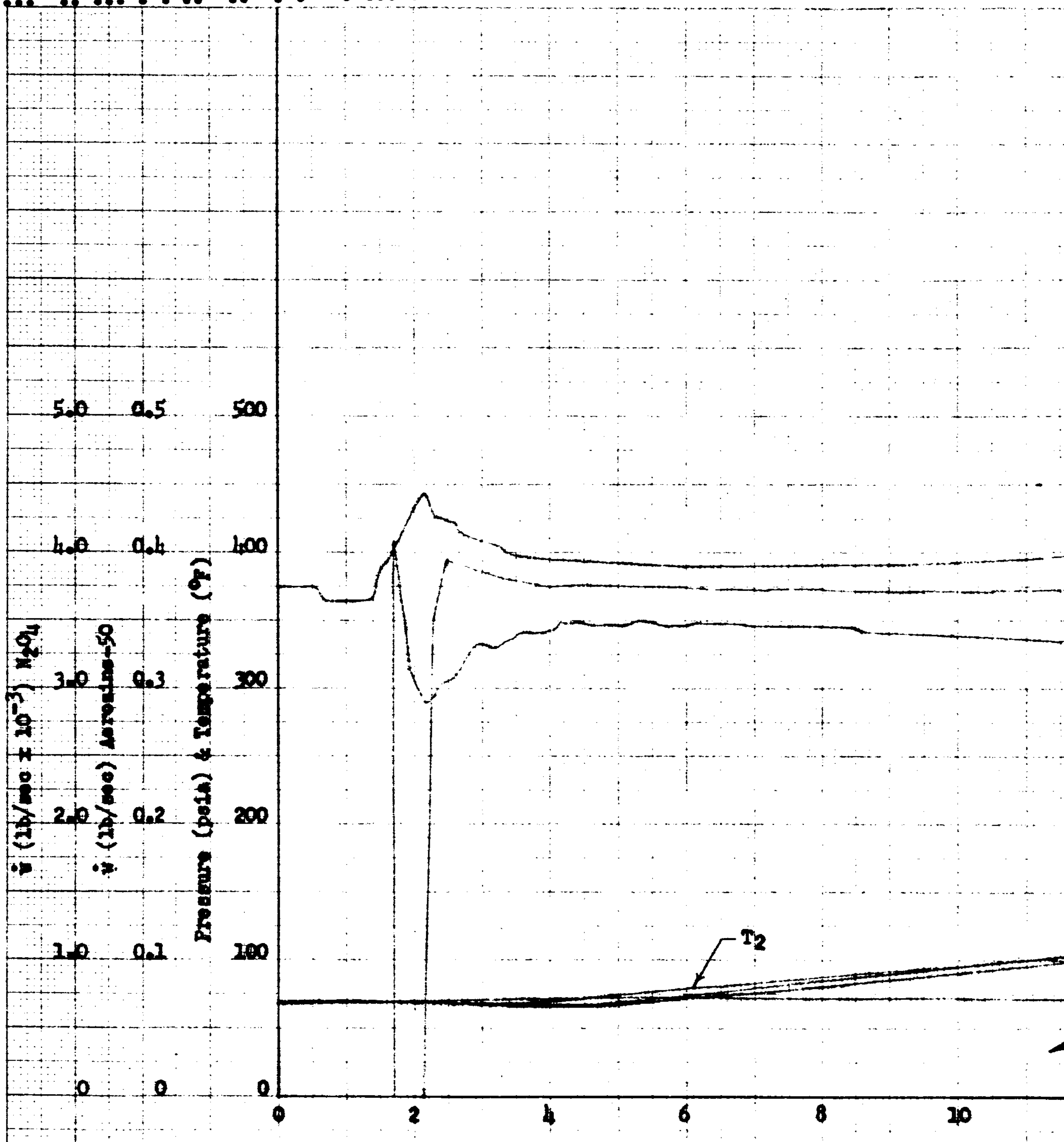

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THIS DOCUMENT CONTAINS INFORMATION [redacted] THE NATIONA[redacted] THE UNITED [redacted] WITHIN THE MEANING OF THE ESPI[redacted] 18, U.S.C., SECTIONS 793 AND 794. THE [redacted] REVELATION OF ITS CONTENTS [redacted] MANNER TO AN UN[redacted] PERSON IS [redacted]ED BY LAW.

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D-6-1

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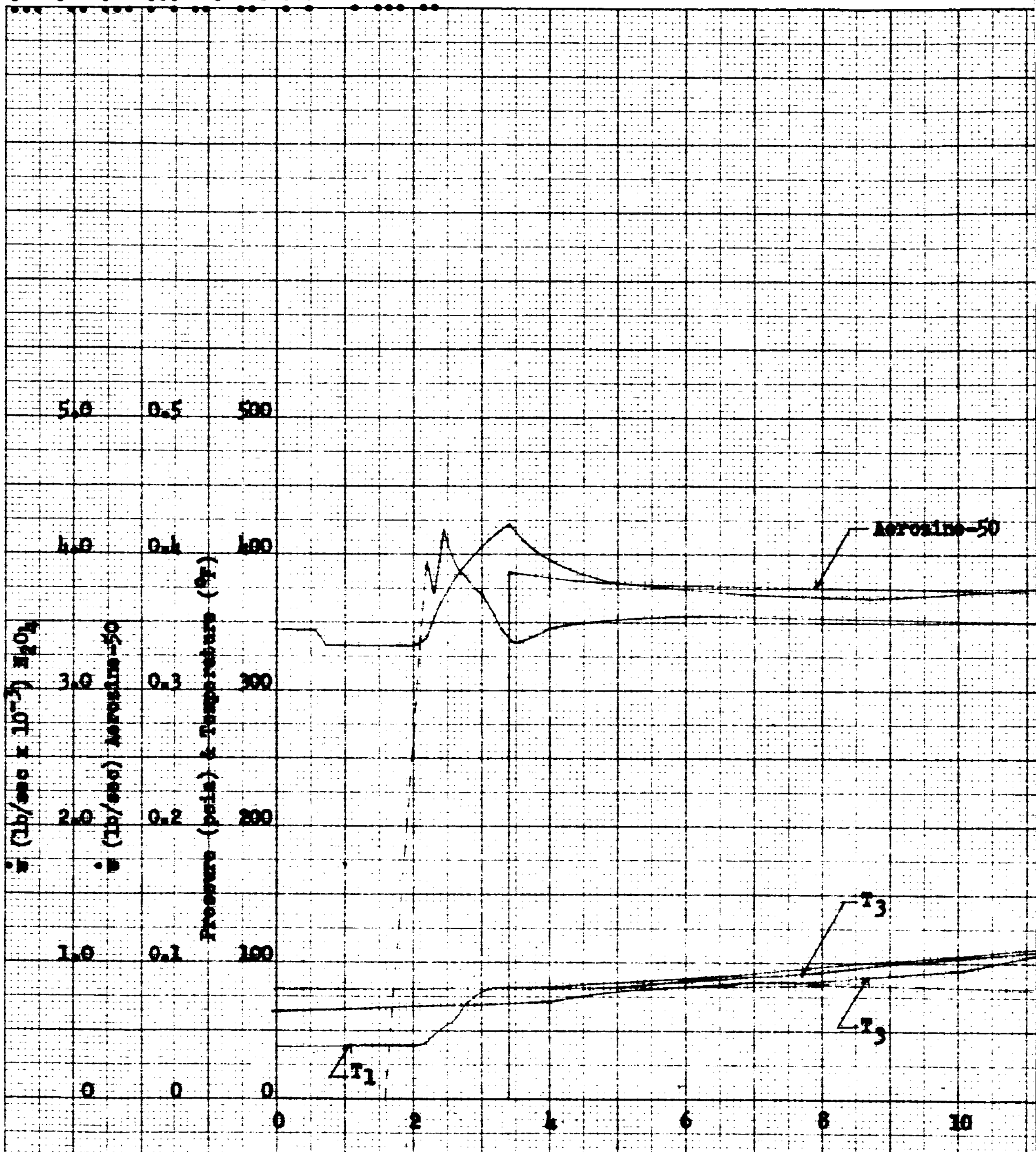
5.0
4.0
3.0
2.0
1.0
0
0.5
0.4
0.3
0.2
0.1
0
500
400
300
200
100
0
• w (lb/sec x 10⁻³) H_2O_2
• w (lb/sec) Aerozine-50
Pressure (psia) & Temperature (°F)
Aerozine-50
T_3
T_5
T_1
0
2
4
6
8
10

1.3 GALLON MAIN TANK INJECTION PRESSURIZATION

N_2O_4 Into Aerozine-50

Bottom Fixed Injector
(1/2" from bottom)

Run 9B 3-19-62

Total Flows

Aerozine-50	N_2O_4
8.62	6.76×10^{-2}

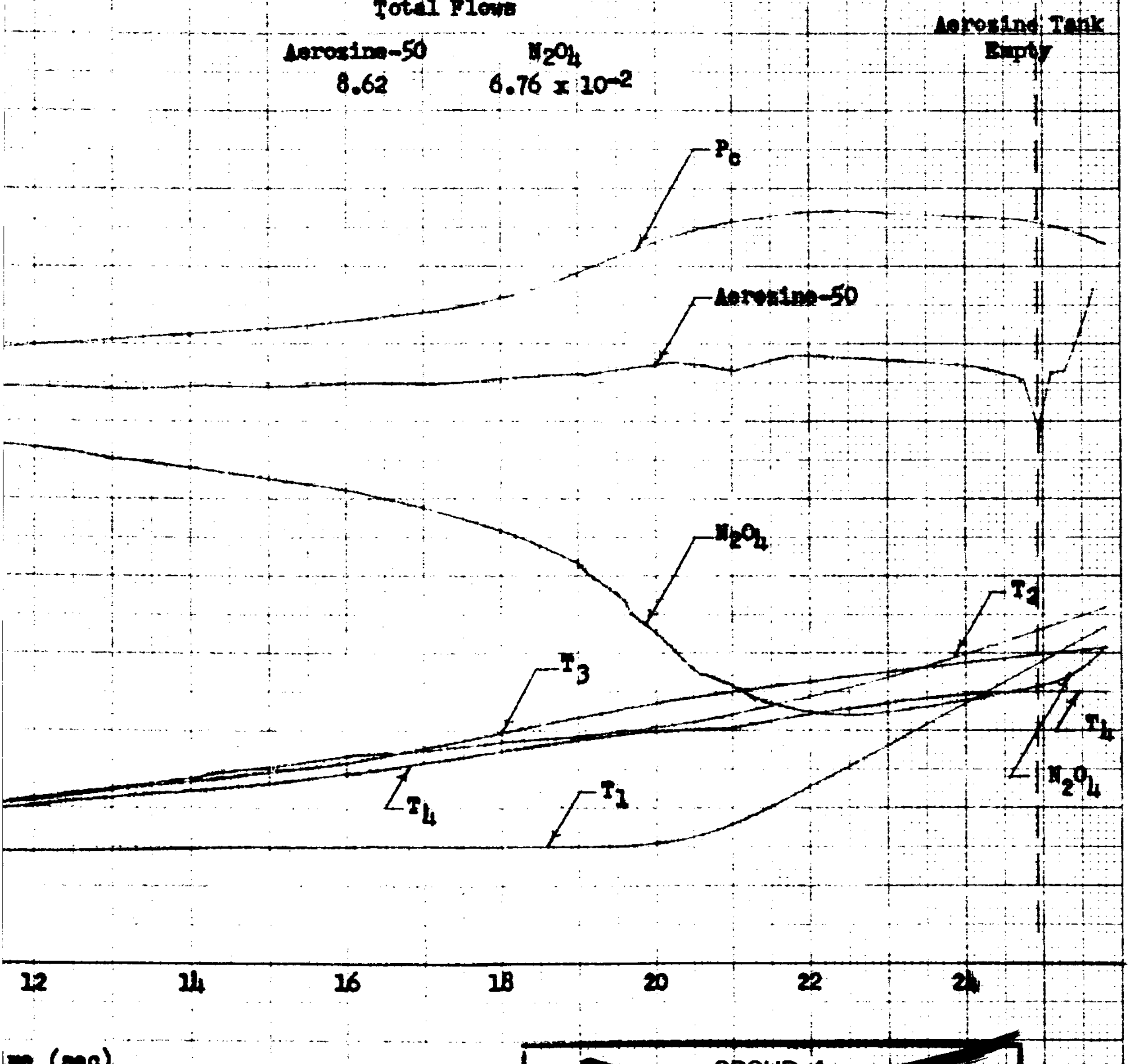

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Figure V-6-2

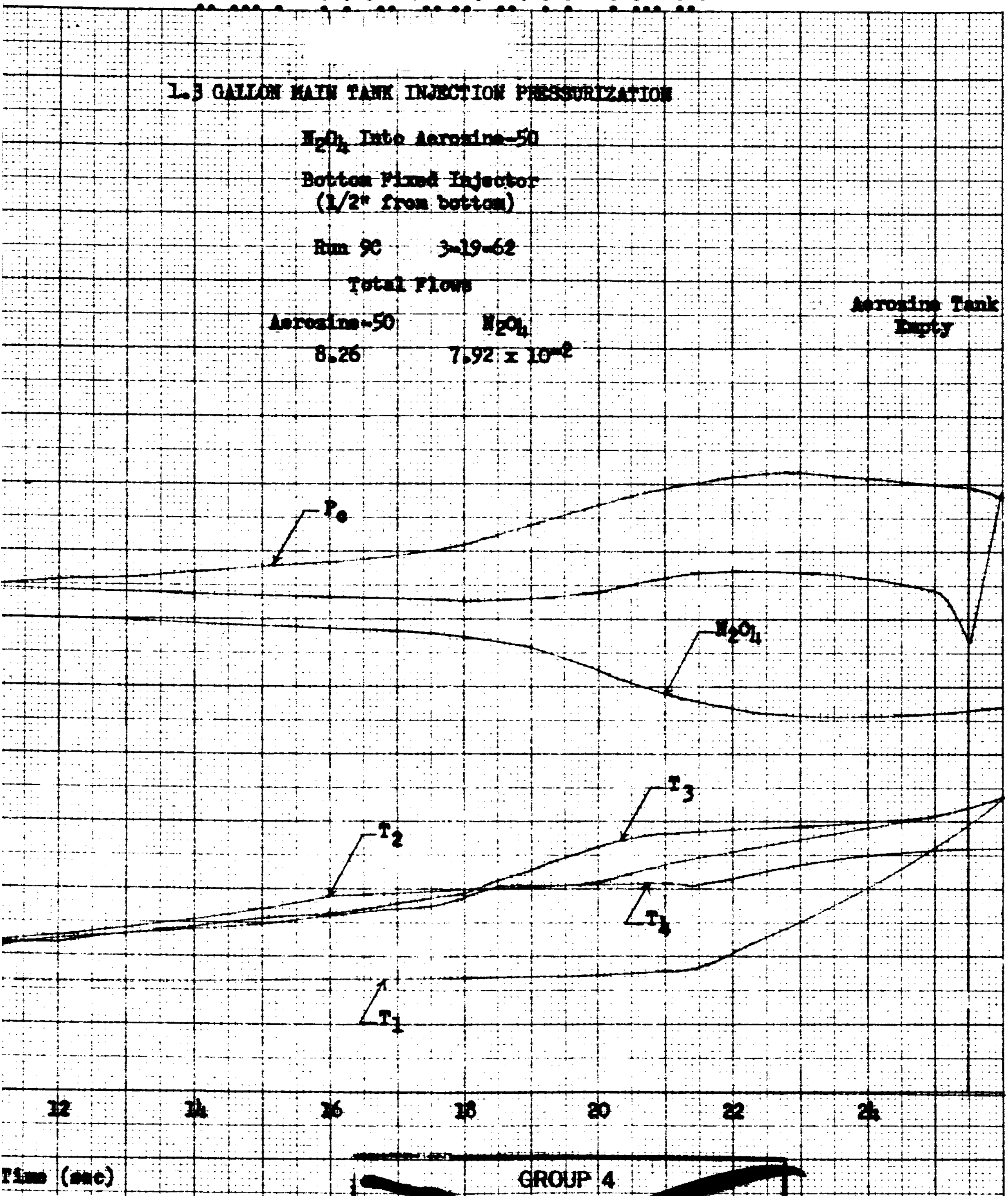

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Figure V-7

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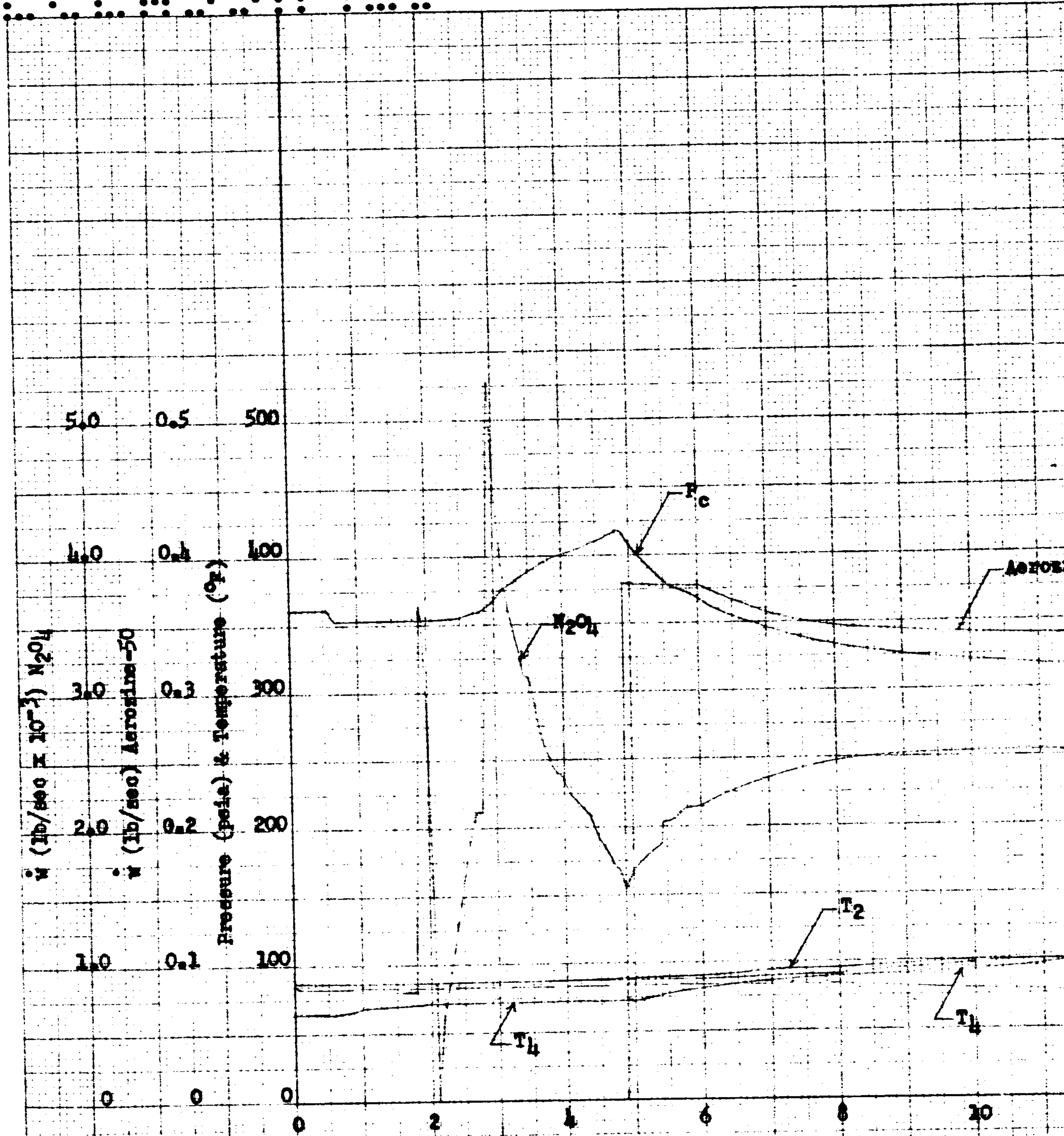

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V-8-1

UNCLASSIFIED

1.3 GALLON MAIN TANK INJECTION PRESSURIZATION

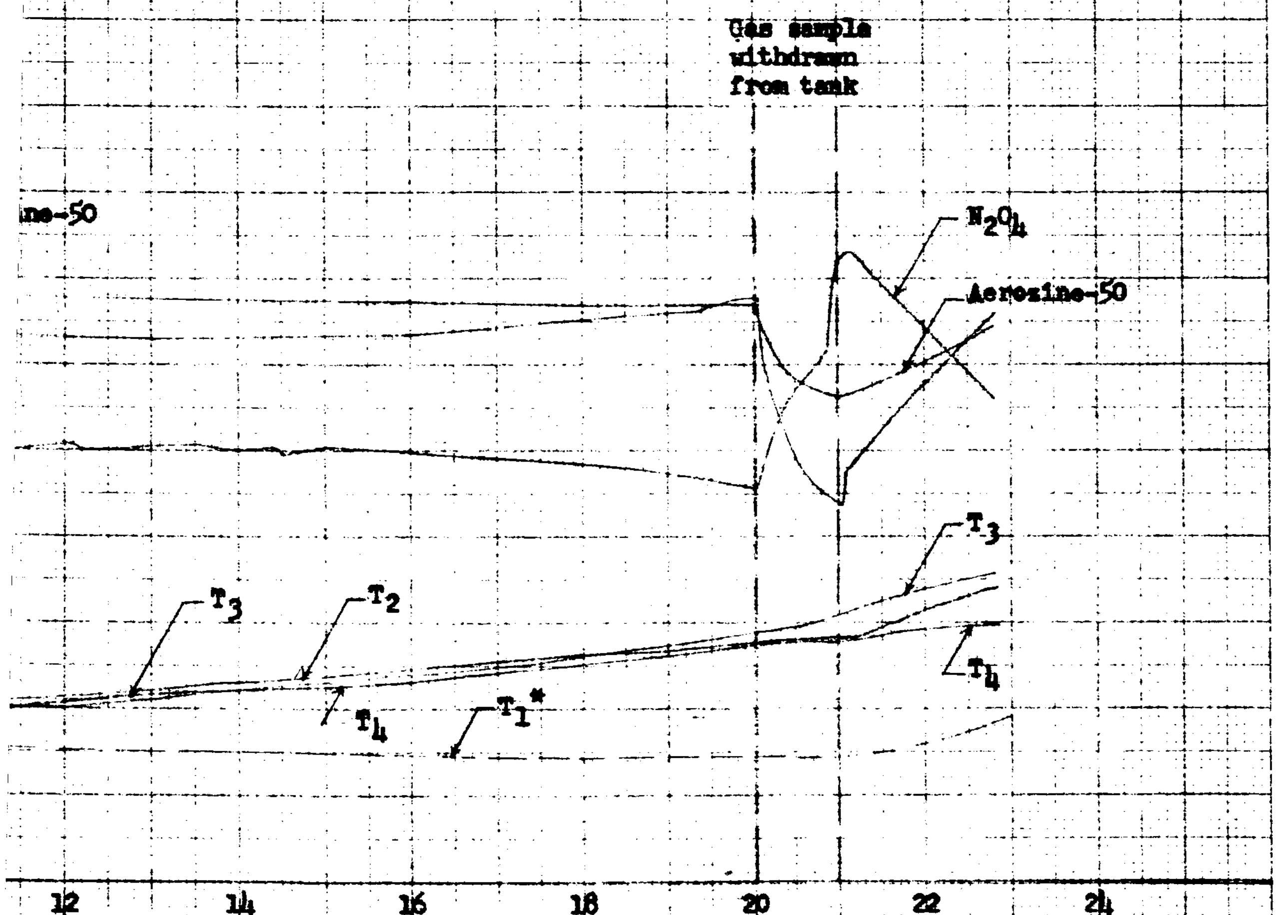
N_2O_4 Into Aerozine-50

Bottom Fixed Injector
(1/2" from bottom)

Run 9D 3-19-62

*Oscillograph trace suggests possible faulty electrical connection on T_1.

Manual shutdown. Aerozine tank not emptied.



Time (sec)

[redacted]

Figure V-8

2

VI. CONCLUSIONS AND RECOMMENDATIONS

A. EVALUATION RESULTS

An evaluation of propellant pressurization systems was performed for two future space missions; a manned lunar landing and return vehicle and an unmanned Mars orbital vehicle. The results of the evaluation showed the most suitable pressurization systems to be those tabulated below.

Lunar Mission	Mars Mission
Monopropellant GG/vaporized fuel	Cross feed main tank injection
Heated helium/vaporized fuel	Main tank injection
Cross feed propellant main tank injection	Monopropellant gas generator
Hybrid gas generator	Hybrid propellant gas generator
Bipropellant gas generator	Bipropellant gas generator

One undeveloped system, the main tank injection system, rated very highly for both missions. This system shows promise of high reliability and light weight.

Very limited main tank injection system testing has been performed with modern propellants; even less work has been done on subsurface injection. It is strongly recommended that a test program be conducted to determine the reactions which take place when various hypergolic propellants are cross-injected. The test work which was performed during this program served to prove the feasibility of subsurface main tank injection of N_2O_4 into Aerozine-50 but many questions remain to be answered: the method of injection, the design of injectors, effects on pressurizing gas density, control of ullage gas temperature, contamination of propellant, and the use of supplementary injectants to produce more desirable reactions.

Proposal SD-62066 has been submitted to the National Aeronautics and Space Administration to produce a detailed description of recommended work in main tank injection.

B. COMPATIBILITY OF PROPELLANTS WITH PRODUCTS OF COMBUSTION

A problem which often confronted us during the evaluation of propellant pressurization systems was the determination of compatibility of propellants with various products of combustion. No handbook or summary was found which lists acceptable limits for reactive gases and propellants. Such a document, listing compatibility and recommending inert gas diluents would prove extremely useful to the aerospace industry.

C. EXPANSION OF SYSTEM EVALUATION

Trial usage of the system evaluation technique was found to be easily workable, while providing an increase in the objectivity of the system selection. The accuracy with which system selection is carried out is dependent upon the number of factors which are included in the evaluations. The present study was limited to an evaluation based upon reliability, weight, size, and cost. It is recommended that the evaluation be expanded to include such factors as:

1. System response time
2. Propellant compatibility
3. Operating pressure, tolerance, and repeatability
4. Logistic considerations
    a. Ease of handling
    b. Checkout requirements
    c. Shelf life
5. Packaging of components

It is further recommended that the search for new and novel propellant pressurization systems be continued. As these new systems are uncovered they can be incorporated into the Design Guide for eventual comparison with the existing systems.

D. EVALUATION BY COMPUTER

The rating and evaluation technique, which is presented in this Report, is structured in a manner which lends itself to computer programing.

Input to the program would consist of two types of information: mission requirements and engine data would provide the input for calculations of system weight, size, etc., and secondly, a selection of the stored influence coefficient curves would be made.

The computer will use the engine data and mission requirements to calculate the propellant pressurization system design criteria. With these criteria, the rating factors for each system will be calculated (reliability, weight, etc.). The computer will then apply these values to the selected influence coefficient curves and compute the final numerical rating of each system. The computer printout will list the systems in the order of suitability for the mission, and the evaluation and selection will have been accomplished rapidly and objectively.

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APPENDIX

DERIVATION OF EQUATIONS IN THE DETERMINATION OF THE PRESSURIZATION UTILIZATION FACTOR

Definition

Utilization factor is defined as the ratio of the inlet gas temperature to the final bulk gas temperature in the propellant tank = $\dfrac{T_g\ (^{o}R)}{T_f\ (^{o}R)}$

Derivation

From the first law of thermodynamics

$$E = W + Q \tag{1}$$

Experimental data indicate that for durations of 50 to 100 sec the tank wall temperature will be quite close to the final bulk gas temperature. Assume that no heat is lost to the surroundings. Therefore, the total heat lost is the heat absorbed by the uniform tank wall in contact with the pressurizing gas.

$$Q = \bar{C}w \quad W_T \frac{(V_o + V_i)}{V_T} \quad (T_f - T_o) \tag{2}$$

The external work done is

$$P_T V_1 \tag{3}$$

The energy change of the pressurizing gas is the enthalpy less the internal energy of the gas after the firing duration.

$$W_1 P_T v_g + W_1 C_{v_g} T_g - W_1 Cv_f T_f = W_1 \quad C_{pg} T_g - W_1 C_{v_f} \ T_f \tag{4}$$

The system originally has a gas at P_o and T_o in the ullage volume. Assuming perfect gas laws allow the number of moles in the tank before and after the firing duration to be calculated as follows:

$$W_o = P_o V_o / RT_o \tag{5}$$

$$W_g = P_T \, (Vo = Vi) /RT_f \tag{6}$$

$$W_o = M_o P_o V_o / RT_o \tag{7}$$

$$W_1 = \frac{M_1}{R} \left[P_T \frac{(V_o + V_i)}{T_f} \qquad \frac{-P_o V_o}{T_o} \right] \tag{8}$$

The total energy change of the gases is the sum of the energy change of the pressurant gas and the energy change of the ullage gases.

$$W_1 \, C_{pg} \, T_g - W_1 \, C_{v_f} \, T_f + W_o \, \overline{C}_{v_1} \, (T_o - T_f) \tag{9}$$

Knowing the energy change, work done, and the heat lost, an energy balance can be written for the system using the first law of thermodynamics.

$$E = W + Q$$

$$W_1 \, D_{pg} \, T_g - W_1 \, C_{vf} \, T_f + W_o C_{vl} \, (T_o - T_f) = \tag{10}$$

$$\frac{P_T V_1}{J} + U_w W_T \frac{(V_1 + V_1)}{V_T} \quad (T_f - T_o)$$

Consider the case of storable propellants. The operating temperatures are 400 to 2000^oR, and the operating pressures vary from 100 to 600 psia. In this range of temperatures and pressures, it is known the C_p and C_v do not change considerably for most pressurizing gases. Therefore, in this analysis it is assumed that $\frac{C_{pg}}{C_{pf}} = \bar{\gamma}_1$ and $\frac{C_{pf}}{C_{vo}} = \bar{\gamma}_o$

The $\bar{\gamma}_1$ is the average specific heat ratio between the inlet temperature and the final bulk gas temperature. The $\bar{\gamma}_o$ is the average specific heat ratio between the final bulk gas temperature and the initial ullage temperature.

Substituting Equation (7) and Equation (8) into Equation (10) and rearranging

$$\frac{P_1 V_o \, (T_o - T_f)}{(\bar{\gamma}_o - 1)\, T_o} + \left[\frac{P_T \, (V_o + V_1)}{(\bar{\gamma}_1 - 1)\, T_f} - \frac{P_o V_o}{(\bar{\gamma}_1 - 1)\, T_o}\right] (\gamma_1 \, T_g - T_f) = \tag{11}$$

$$P_T V_1 + \bar{C}_{wJ} \, W_T \, \frac{(V_o + V_1)}{V_T} \, (T_f - T_o)$$

$$\text{where } \frac{1}{\gamma - 1} = \frac{JMCv}{R}$$

The left-hand side of Equation (11) becomes

$$P_T \, \frac{(V_o + V_1)}{\bar{\gamma}_1 - 1)} \left[\frac{\bar{\gamma}_1 \, T_g}{T_f} - 1\right] + \frac{P_o V_o}{T_o} \left[\frac{T_o - T_f}{(\bar{\gamma}_o - 1)} - \frac{T_g - T_f}{(\gamma_1 - 1)}\right] \tag{12}$$

Dividing both sides of Equation (11) by $\dfrac{P_T \, (V_o + V_1)}{(\bar{\gamma}_1 - 1)}$

$$\bar{\gamma}_1 \, \frac{T_g}{T_f} = 1 + \bar{\gamma}_1 - 1 \left\{ \frac{V_1}{V_o + V_1} + \frac{\bar{C}_w J \, W_T}{V_T P_T} \, (T_f - T_o) + \frac{P_o V_o}{P_T \, (V_o + V_1)} \left[\frac{\bar{\gamma}_1 \, T_g - T_f}{(\bar{\gamma}_1 - 1)\, T_o} + \frac{T_f - T_o}{\bar{\gamma}_o - 1)\, T_o} \right] \right\} \tag{13}$$

The last term is usually small, and hence to eliminate T_f from it the assumption is made that $\bar{\gamma}_1 - 1 \approx \bar{\gamma}_o - 1$. This gives

$$\frac{T_g}{T_f} = \frac{1}{\bar{\gamma}_1} + \frac{(\bar{\gamma}_1 - 1)}{\bar{\gamma}_1} \left\{ \frac{V_1}{V_o + V_1} + \frac{\bar{C}_w J \, W_T}{V_T \, P_T} \, (T_f - T_o) + \frac{P_o V_o (\bar{\gamma}_1 \frac{T_g}{T_o} - 1)}{P_T (V_o + V_1) \, (\bar{\gamma}_o - 1)} \right\} \tag{14}$$

The Equation is subject to the limitations and assumptions made in its derivation.

LIST OF SYMBOLS

C_{pg}	Specific heat of pressurization gas at T_g
C_{vo}	Specific heat of ullage gas at T_o
$\bar{C}_{v1}$	Specific heat of pressurization gas between T_o and T_f
$\bar{C}_w$	Specific heat of tank material between T_o and T_f
C_{vf}	Specific heat of pressurization gas at T_f
ΔE	Total change of internal energy
J	Mechanical equivalent of heat
M_o	Molecular weight of ullage gas
M_1	Molecular weight of pressurization gas
P_o	Pressure in tank before firing
P_T	Tank pressure during firing
Q	Total heat lost
R	Universal gas constant
$\bar{\gamma}_o$	Average specific heat ratio of ullage gas between T_o and T_f
$\bar{\gamma}_1$	Average specific heat ratio of pressurization gas between T_g and T_f
T_o	Temperature in tank before firing
T_g	Inlet temperature of gas into propellant tanks
T_f	Temperature of pressurizing gas after firing
v_g	Specific volume of pressurization gas at T_g
V_o	Ullage volume before firing
V_1	Volume of displaced propellant
V_T	Total tank volume
W	External work done by gas
W_o	Weight of ullage gas
W_1	Weight of pressurization gas
W_g	Total weight of gases in tank
W_T	Weight of tank material